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Preliminary Design Report 2
and
the Final Report

A Manned Mars Mission

Spacecraft Design Group
University of Texas
at Austin

May 1985

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EXECUTIVE SUMMARY

The NASA administration has requested Johnson Space Center (JSC) to outline NASA's long term objectives for the nation's manned space program. In developing these long term objectives, JSC is emphasizing the development of a rational long range plan, objectives which are compatible with current and near term programs, a theme of permanent manned presence in space, and an effort towards foriegn cooperation in space.

There are several reasons why the manned exploration of Mars is a logical long term goal for the manned space program. Mars is the most earth-like of the other planets of the solar system. The scientific exploration of Mars is a major objective of the solar system exploration program. Mars and its moons are a reasonable transportation node for launches to other places in the solar system with the use of in-situ propellent production. There are also several opportunities for international cooperation in the exploration of Mars.

JSC wants to study the opportunities for manned exploration of Mars in a systematic way. The long term program directions and requirements must be defined. The impact and requirements on current and near term programs must be determined, and key technology areas must be developed.

The objective of this group is to design a mission to Mars which will contribute to NASA's long term objectives. The group will design a primary mission concept which will land men on Mars and contribute to the long term exploration of the planet. This project will involve two preliminary design reviews, and a final report, and will be completed by the end of the 1985 spring semester. The University of Texas at Austin (UT) will perform all the mission planning for the interplanetary section of the mission and design all vehicles and mission planning for the exploration of Mars and it's moons. UT will coordinate it's efforts with Texas A&M University's design team, who will design the major portions of the interplanetary vehicle(s).

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1.0 GENERAL SUMMARY

The intent of this project is to plan a manned mission to Mars. This mission will include a manned and unmanned exploration of Mars. It is assumed that the mission will begin at a low Earth orbit Space Station. It has also been assumed that the exploration of Mars will be a long term goal rather than a short term one, and that many manned missions will go to Mars and develop permanent habitats. The first manned landing on Mars must logically contribute to the missions that will follow it. The vehicles proposed in this report will be designed to be applicable to future space exploration.

The design group for this project was divided into two subgroups. The first subgroup is the Mission Planning group and the second is the Martian Operations group. The responsibilities of the Mission Planning group include of trajectory analysis, planetary capture, vehicle payload configuration, interplanetary vehicle design, and Martian Moon exploration. Trajectory analysis is concerned with studying a ballistic and a low-thrust trajectory to Mars from a low Earth orbit. The methods of planetary capture that are proposed are aerobraking, aerocapture, propulsion, and a combination of propulsion and drag braking. There are two proposed vehicle payload configurations. The first suggests two vehicles, one for cargo and the other for the crew. The second has a single vehicle which carries everything. Vehicle design is a study of the requirements and problems of designing an interplanetary vehicle. Finally, the Martian Moon exploration proposes either docking the interplanetary vehicle with the moons or using remote probes for surface exploration of the moons.

The responsibilities of the Martian Operations group include design of an ascent/descent vehicle, a Mars "rover", a laboratory/habitation unit, and unmanned probes. The Martian Operations group will also develop an operational plan which describes the manned and unmanned surface exploration scenario. The preliminary design of these structures has been completed.

The ascent/descent vehicle will land at two different sites. The ascent portion will be a multiple use vehicle, while the descent portion will be a disposable, one-use vehicle. An ascent/descent package will land at Site 1 and serve as a base of operations. Then the ascent vehicle will return to the orbiting vehicle, connect to another descent section, and land at Site 2.

The Mars surface mobility vehicle, termed the "Rover" in this paper, is designed to house two astronauts in a shirt-sleeve environment. The Rover will use loopwheels to travel the distance between Site 1 and Site 2.

The laboratory/habitation unit, or Lab/Hab, will provide a scientific laboratory and living quarters for four astronauts. The Lab/Hab will act as a base of operations for the astronauts at Site 2.

Two different types of unmanned probes will be used to explore the surface of Mars, Phobos, and Deimos. Penetrators will be deployed to Mars and a non-recoverable Soft Lander to the moons.

2.0 INTERPLANETARY OPERATIONS

The Interplanetary Operations group is primarily concerned with transporting equipment and supplies to Mars. Areas of analysis include trajectory, capture, and vehicle design.

In order to maximize payload capabilities and minimize time of flight for the astronauts, the Interplanetary Operations group has developed a mission plan consisting of a two vehicle team. The lead vehicle is a large, unmanned supply vehicle, or "barge", which will be used to transport payload items such as exploration vehicles, scientific equipment, and replenishable supplies. Departing 1-3 years before the manned vehicle, the barge will use a fuel-saving low thrust propulsion system. The manned vehicle will be designed for rapid transit and will use either ballistic propulsion or a high acceleration low thrust propulsion system. The two vehicles will rendezvous in Martian vicinity and begin the exploration phase of the mission.

2.1 VEHICLE DESIGN

The Vehicle Design Group was responsible for developing a workable barge configuration and summarizing barge system requirements. A vehicle layout has been completed and required systems have been defined.

2.1.1 Barge Configuration

The vehicle layout is illustrated in Figure 2.1.1.1. This illustration has been drawn to scale and all major components are indicated.

The propulsion system is a mercury ion engine that is comprised of a nuclear reactor, a power conversion unit, and an ion engine/thruster. The reactor uses lithium-7 as a coolant. The coolant loops are placed in planes parallel to the thrust axis and may be automated so that all cooling takes place in the planes with the least solar irradiation. The reactor is housed by a shadow shield made of a high density material. This shadow shield casts a protective cone so that docking vehicles can approach the barge without contamination from the propulsion system. Detailed engine specifications are listed in Table 2.1.1.1.

Mercury will be stored as a liquid in the four double-walled containers at the aft end of the barge. These double-walled containers provide micrometeorite protection. The tanks will be sized so that failure of up to two tanks will not threaten mission success. Allocating twice the required amount of fuel both increases probability of mission success and adds versatility to the overall mission.

Three ring structures are positioned along the vehicle core as illustrated in Figure 2.1.1.1 The aft ring houses all observation equipment, flight control systems, guidance systems, and communication equipment. Boom-mounted antennae will be deployed and attached to the aft ring for data transfer and guidance control. The middle ring houses all unmanned probes and penetrators. The forward ring serves as an environmental vestibule. The vestibule allows access to the A/D vehicles and is connected to the environmental core section that extends to the docking interface. The life support system for the vestibule ring and the core will be contained on the barge.

The environmental vestibule is equipped with six airlocks and accompanying docking mechanisms for access to the ascent/descent vehicles. These are located in pairs sixty degrees apart and are shown in side view in Figure 2.1.1.1 The ascent/descent vehicles will be docked with the three axial airlocks in transit to Mars and will be used for debarkation to the planet's surface. The three perpendicular airlocks will be used in orbital rendezvous with the ascent vehicles. One of these is included for redundancy. An airlock is also located at the forward end of the environmental core for docking with the manned interplanetary vehicle.

The core is the central element of the vehicle structure. The core must also integrate all of the components and component systems. One of the major problems associated with a long thin structure is buckling. Buckling results from structural instability under axial loading conditions. Another problem with a structure of this length is bending moments that will be developed from transverse forces. Both the buckling and the bending problems must be analyzed when sizing the core cross section.

A simplified buckling analysis was performed to determine a core wall thickness. The material properties of a solution treated aluminum alloy were used in this analysis. This material was selected on the basis of:

1. specific strength
2. specific stiffness
3. magnetic properties
4. availability

The nominal inside diameter of the core section is 72 inches. With this inside diameter and considering the thrust from the nuclear propulsion system, the outside diameter was found to be 82 inches. With these dimensions, the hollow cylindrical core has the following characteristics:

Moment of Inertia:	9.0×10^5 in.^{4}
Area:	1.21×10^3 in.^{2}

A free-free beam analysis was performed and the maximum bending moment due to a transverse force applied at the end was determined. The moment of inertia requirements for this loading condition were found to be much less than for the the axial loading condition. Therefore, the buckling criteria was used in the final size and weight estimation of the core.

2.1.2 Barge Systems

Many systems are required for navigation, guidance, and rendezvous operations. Because these systems must be integrated with the rest of the vehicle, they often have considerable effect on vehicle design. Hence, the following vehicle systems will be covered in this section of the report:

1. Thermal Balance
2. Reaction Control
3. Navigation/Guidance/Telecommunication
4. Multi-vehicle Interface
5. Equipment Status Monitor
6. Power Distribution

These systems must work together to protect, control, and maintain the barge and interact with the environment.

2.1.2.1 Thermal Balance Subsystem

The four major variables that will affect the temperature of the spacecraft components will be incident solar radiation, electrical power expenditure, thermal transfer between components, and thermal radiation from the spacecraft into space. Many components of a spacecraft operate within narrow temperature limits. The thermal balance system is comprised of all devices and treatments employed to maintain temperatures

within specified bounds by achieving a proper balance between absorbed and radiated heat. The principle active temperature control devices are variable-emittance louver assemblies. They are used primarily around the housing that will contain the vital electrical and electronic assemblies. Passive devices include multi-layer aluminized-mylar blankets, polished aluminum shields (for electronic assemblies), and surface treatments having particular thermal properties.

The solar absorptivity constant (SAC) indicates how much solar energy will be emitted or absorbed. Given the SAC and the solar constant (SC), the Stefan-Boltzman Law can be used to compute the surface temperature of an object. Components requiring reflection of solar radiation need an SAC of about 0.1 to 0.3 . This is achieved by coating the components with silicone elastomers pigmented with titanium oxide. Component requiring solar radiation absorption must have an SAC of about 6. This can be accomplished by combining bare metal layers and layers coated with dark paint.

2.1.2.2 Reaction Control System

The reaction control system (RCS) consists of 24 reaction control thrusters. Every thruster location and direction will be duplicated to provide control system redundancy. The aft thrusters are to be positioned on booms in order to minimize plume impingement and "blowback" contamination during accelerated or decelerated flight.

The RCS performs several important functions:

1) The RCS controls attitude and low thrust direction.
2) The RCS adds translational manueverability to the spacecraft.
3) The RCS offers stability for navigational sightings and scientific measurements with directional dependency.

A fourth possible use of the RCS is active structural damping. As sensors located throughout the structure detect vibration and stresses, the RCS thrusters could respond with corrective thrusting. The result would be a more stable structure with minimal stresses. These are important considerations for large space structures.

The amount of RCS propellant needed on the barge will depend on how extensively the RCS is used for each of the functions defined above. A general estimate can be made based on the projected energy expenditure of

2 million lb-sec per year for the Space Station. The barge will probably have an energy expenditure on the order of 6 million lb-sec for its 2-4 year mission.

The RCS propellant could be based on the hypergolic bi-propellants nitrogen tetraoxide (oxidizer) and monomethyl hydrazine (fuel). Both are capable of existing as liquids in 20° to 100° C. With a specific impulse of about 315 seconds, this fuel could be used to produce 900 lb. of thrust. Using the estimated 6 million lb-sec energy consumption, approximately 19,000 lb of fuel would be required. With a design mixture ratio (oxidizer weight to fuel weight) of 1.65, four 33 cubic foot tanks would be able to store the 19,000 lb of fuel.

2.1.2.3 Guidance/Navigation/Telecommunication Systems

The guidance subsystem will be used to implement course corrections initiated by navigation sensors and instruments. Upon arrival at Mars, these systems will be used to maintain the barge in the desired orbit and attitude. Some of the following instruments and sensors might be used for guidance, navigation and attitude control: star trackers, sun sensors, horizon sensors, rate gyros, accelerometers, and inertial measurement units. A centralized data processing unit could be used to disseminate incoming information and communicate with the reaction control system.

The barge telecommunications system should consist of radio receivers, transmitters, antennas, data encoders, and video storage. The system must be able to send and receive all command, control, telemetry, and general communications.

2.1.2.4 Multi-vehicle Interface System

Various mission scenarios require rendezvous and docking maneuvers. For rendezvous and docking operations, instruments and sensors for the docking/tracking system will be required. The following specifications offer approximate performance standards.

Maximum range of operation	100 km	
Range Tracking Accuracy	10 - 200 m	: (+/-) 2%
	200 - 100,000 m	: (+/-) 5%
Range Rate Accuracy	10 - 10,000 m	: (+/-) 0.15%
	10,000 - 100,000 m	: (+/-) 5 %
Angle Accuracy	10 - 20 m	: (+/-) 1°
	20 - 100,000 m	: (+/-) 0.5°

An automatic docking system could be used in conjunction with the onboard tracking system to guide unmanned vehicles to the barge via commands sent from the barge. The automated system might consist of a remote manipulator arm (RMA) and a central command station. The central command station would interface with the barge tracking system and the reaction control systems onboard the docking vehicles. As the tracking system monitors relative changes in position and orientation, appropriate reaction control system commands could be transmitted to the approaching vehicle. The central command station could also control the barge orientation with commands to the barge RCS. Portions of the automated system, such as the RMA, could also be used to perform docking operations with manned vehicles.

2.1.2.5 Equipment Status Monitor System

Many of the subsystems transported to Mars on the barge will not be used until the manned vehicle arrives. In order to insure a successful mission, the operational status of these subsystems must be known. Therefore, an independent subsystem should be used to continually monitor the operational status of the onboard equipment

2.1.2.6 Power Distribution System

A generous supply (ie: Mega Watts) of electrical power will be available from the nuclear power generation system being designed by Texas A&M University. This should be more than enough to satisfy the

power requirements of the barge subsystems, which will probably be on the order of 100 KW.

The onboard circuitry will be controlled by a centralized power distribution module.Some barge system components will always receive power while others will be turned on and off during various parts of the mission. Direct current to alternating current inverters might be used to generate an aerospace-standard 400 Hz, 28 Vrms power.

2.2 Moon Exploration

Two concepts are being considered for moon exploration:

1) Docking the barge on the moons
2) Exploring the moons with unmanned probes.

Before choosing a moon exploration scenario, an optimal parking orbit will be chosen. This choice will be based on the total momentum change required for each scenario. By this method of analysis, we hope to minimize the total energy expenditure of the for this portion of the mission. Our present analysis assumes a parking orbit of 270 nm (500 km). More analysis is being done to finalize the parking orbit altitude as well as determining structural requirements for landing.

The menu of choice landing sights allows orbit inclinations to vary from -5 to +5 degrees from the 0 degree equatorial orbit. Since the orbit inclination of Phobos and Deimos is approximately 1.8 and .9 degrees respectively, plane changes may be necessary in moon rendezvous and docking operations. The delta v cost of a plane change is an important mission consideration. Fig. 1 shows how the delta v requirement for moon rendezvous is effected by a variation in parking orbit inclination. These particular delta v's are based on a Hohmann transfer from a 270 nm. parking orbit. As would be expected, the minimum delta v's occur when parking near the inclination of one of the moons. The maximum delta v occurs when coming from a -5 degree inclination, the negative limit of our preset range of parking orbits. Yet, even for that extreme case, the total delta v requirement is only 1% to 3% higher than it would have been without a plane change.

To achieve docking, the barge will perform a Hohmann transfer to the target moon until the vessel is within 100 m of the moon. At that point, the vessel will match velocities with the moon and then free fall to the moon's surface. The velocity of the vessel at impact will be 1.1296 m/s for Phobos and .8567 m/s for Deimos. A bumper made of collapsible honeycomb material can be used to absorb the impact. Cables will be used to anchor the barge to the moon. Three spools of cable will be mounted at the top of the barge. When the vehicle has landed, small rockets attached to the end of each cable will fire and implant each cable into the moon's surface. The spools will then reel in the cables until they are taut. Since we know very little about the composition of the moon, alternate implantation methods

should be designed to penetrate a variety of surfaces. An unscaled drawing illustrating this basic concept can be seen in Figure 2.2.1.

2.3 TRAJECTORY ANALYSIS

2.3.1 BALLISTIC TRAJECTORIES

Introduction and Theory

This section of the report is concerned with calculating a ballistic trajectory from Earth to Mars, using the patched conics orbital transfer method. A ballistic trajectory is one in which the spacecraft is given an impulsive change in velocity at departure and arrival and allowed to coast during the interplanetary phase of the mission. Thus, the trajectory is divided into three separate conic sections. When the spacecraft is near the Earth's sphere of influence (approximately 621,500 miles or less), the orbit is modeled as one conic section. Beyond this sphere of influence, the motion of the spacecraft is governed by the sun's gravitation. This phase is the second conic section. The procedure used to determine the trajectory of this section is Lambert targeting. Finally, the orbit of the spacecraft inside Mars sphere of influence is the third conic section. The initial condition for the first phase was that the spacecraft be in a circular 270 nautical mile parking orbit around the Earth. Then, the final conditions from the first phase (position and velocity) are used as the initial conditions for the second and so on; hence the term patched conics. The final condition for this mission was that the spacecraft be in a 270 n. mi. parking orbit around Mars. The criteria used for determining the optimal ballistic trajectory using this method are: (1) energy requirements, (2) time of flight, and (3) launch windows.

Discussion

The energy required by the spacecraft to make a ballistic transfer from Earth to Mars is directly proportional to the delta-V the vehicle makes. Thus, the minimum energy interplanetary transfer is a Hohmann transfer ellipse, in which the craft rendezvous with Mars 180 degrees away from the launch point. For this maneuver to take place, Mars must be leading Earth by the proper phase angle, which turns out to be 44.4 degrees. Fig. 2.3.1.1 shows a plot of the phase angle for various launch dates. From this, it can be seen that the correct phase for a Hohmann transfer occurs in November 1998 and December 2001.

The disadvantage of a Hohmann transfer is that the time of flight (TOF) is greater than if a faster trajectory where the craft reaches Mars in less than 180 degrees is used. The TOF for a Hohmann transfer is 260 days. If a faster TOF is desired, the delta-V requirement is correspondingly greater. For this mission, it was determined that a transfer time of 220 days would still keep the delta-V requirements within reasonable limits, so it was chosen as the baseline from which other calculations would be made. Thus, from this baseline, the delta-V's needed for escape and capture have been determined. Also, a plot of the launch dates vs. delta-v (Fig. 2.3.1.2) was made so that a suitable launch date could be found. This launch date in turn shows the phase angle for Mars from Fig. 2.3.1.1.

Results

For the Earth to Mars ballistic trajectory, the following mission profile has been adopted.

Planet departure and capture parameters:

Earth Departure -- circular orbit altitude of 270 n. mi.
Mars Capture -- circular orbit altitude of 270 n. mi.

Velocity Requirements:

Earth escape delta-v = 11520 fps
Mars capture delta-v = 2110 fps
Total delta-v = 13630 fps

2.3.2 Low Thrust Trajectories

For the ballistic trajectories discussed so far, the vehicle would utilize chemical rocket engines to provide impulsive velocity changes to achieve the desired trajectory. The precursor vehicle, however, will employ a low thrust propulsion system in which electrically charged Mercury particles will be accelerated by electric or magnetic fields to attain exhaust velocities much higher than chemical rockets. Although the thrust to mass ratio will be very low (typically on the order of 10^{-5} to 10^{-3} g's), the engine will be capable of delivering this thrust for very long periods of time.

Theoretical Development

Three modes of flight were considered for low thrust trajectory analysis:

1) Spiral out of LEO to just beyond the Earth's sphere of influence

2) Transfer from Earth to Mars after having achieved Earth escape velocity.

3) Spiral into Mars orbit from a radial distance of 135,000 nmi.

For these modes, the equations of motion in vector form can be written as:

$$r + u/r^3 = T/[m_o - m(t-t_o)]$$

where: r = the distance from the spacecraft to the central body (Earth, Sun, or Mars)
u = the gravitational parameter of the central body
T = The magnitude of the thrust; 900 lb_f
m_o = the initial spacecraft mass 2×10^6 for all modes
m = the mass flow rate; 0.00154 lb_m/sec

Assumptions

For a first attempt at generating low thrust trajectories, several assumptions were made to simplify the analysis. The vehicle was treated as a point mass with the thrust and the gravitational attraction of the central body as the only external forces acting on it. All motion took place in the same plane (i.e. no plane changes were taken into consideration). Launch

window considerations were not taken into account. The equations were numerically integrated using a Runge-Kutta 4th and 8th order integration schemes given a set of initial conditions. Following is a list of conditions for each of the three trajectory modes.

Mode 1: Spiral out of Earth parking orbit

a) Circular parking orbit altitude of 200 n.mi.
b) Thrust vector perpendicular to radius vector
c) Numerical integration procedure terminated when spacecraft reached escape velocity.

Mode 2: Earth to Mars transfer

a) The orbits of Earth and Mars are assumed to be circular.
b) Initial spacecraft velocity equal to the Earth's orbital speed (97740 fps).
c) Initial spacecraft radius from the Sun is the same as the Earth's.
d) Thrust vector vector perpendicular to the radius vector.
e) Integration procedure is terminated when the spacecraft reached orbital radius of Mars.

Mode 3: Spiral into Mars orbit

a) Initial velocity just below escape velocity of Mars (1312 fps) with the velocity vector perpendicular to the radius vector.
b) Initial radius equal to 134990 n.mi.
c) Thrust vector always pointing in the direction opposite to the direction of the total velocity vector. Thus will vary with time.
d) Integration terminated when spacecraft reached circular parking orbit altitude (270 n.mi.)

Results

mode 1:

a) time of flight 177 days
b) escape velocity of 3338 fps at radius of 415770 n.mi.
c) 236000 lb_m of propellant consumed

mode 2:

a) time of flight 196 days
b) magnitude of velocity vector upon reaching Mars: 89630 fps
   radial velocity: 31010 fps
   tangential velocity: 84110 fps
c) 261340 lb_m of propellant consumed
d) phase angle of approximately 169 deg.

mode 3:

a) time of flight 59 days
b) speed at 270 n.mi. altitude: 9190 fps (note: required speed for a circular orbit at this altitude is 10928 fps)
c) 78670 lb_m of propellant consumed

Thus, the total trip time is 432 days and to total amount of propellant consumed is 576,010 lb_m.

Recommendations

Since space missions, in general, require the use of great amounts of energy and long travel times, the necessity of performing guidance maneuvers in an optimal manner is quite important. Further analysis should be done to create an optimization scheme to solve a boundary value problem dictated by launch windows and arrival velocities. These analyses should hopefully lead to minimum energy and propellant expenditures to help lower initial mass and therefore enable larger payloads to be carried out to Mars. A complete analysis should also take into account the ellipticity and non-planar motions of the spacecraft, Earth, and Mars. Also, it should be noted that the analysis carried out, treated the three modes as seperate entities where the computations for one mode did not require the results of another. Thus, the next step would be to integrate the three modes together in a "patched conics" type approach where the final result of the first mode would be used as the initial condition for the second, and so on. Finally, with the simplified approach used in the model, it would be almost impossible to set the initial conditions such that the desired parking orbit at Mars could be achieved using low thrust engines. Thus, either a propulsive manuver must be used or a more complicated analysis using optimization techniques must be performed.

2.4 Planetary Capture

A crucial part of the manned mission to Mars will be to achieve a parking orbit about Mars. The transfer from an interplanetary trajectory to an elliptical or circular orbit about a planet is referred to as planetary capture.

The study of planetary capture techniques for the Mars mission has proceeded in two phases. The first of these phases involves an extensive survey of technical literature and related textbooks on the subject of planetary capture. The second phase, which is still in progress, involves numerical analysis, both by hand and with the aid of a computer.

Several methods of capture were considered:

- o Aerocapture
- o Low thrust Spiral
- o Impulse Thrust
- o Aerobraking

Aerocapture obtains the necessary delta-v to achieve parking orbit by passing deep into the Martian atmosphere to dissipate excess energy. Preliminary analysis suggested that an aerocapture attempt at Mars would be too hazardous to warrant its consideration as a viable option for planetary capture; consequently, the aerocapture method was dropped.

Spiralling capture decelerates with low, constant thrust over a great length of time to achieve the delta-v needed for parking orbit. Preliminary analysis showed the spiralling capture to be prohibitive from a temporal stand point, so this option was disregarded. Of the two remaining options, the simpler and more well-known method is propulsive capture. The delta-v needed to enter parking orbit is attained through a single propulsive burn at the proper instant. With the total mass before the burn (m_0), the type of fuel (for value of specific fuel thrust, I_{sp}), and the delta-v all known, the amount of propellant can be easily computed with the equation:

$$m_f = m_0 \exp(\text{delta-v}/(g_0 I_{sp}))$$

where g_0 is the acceleration due to Earth's gravity (a constant), and m_f is the total mass before burn less the propellant mass used for the burn. Table shows the propellant weight for a 1.5 million lbs. spacecraft using a

propellant with an I_{sp}=350 sec. The delta-v was estimated to be 11,644 ft/sec and a 270 nm. circular orbit was assumed

Aerobraking involves successive passes through the atmosphere to slowly degrade the orbit and dissipate excess energy until parking orbit is reached. The advantage of aerobraking over propulsive capture is the smaller fuel requirement, allowing space for more payload. The most important disadvantage of aerobraking is the excessive time needed to complete capture.

Aerobraking at Mars begins with an initial delta-v (delta-v1) of approximately 3080 ft/sec, achieved by a propulsive maneuver at a radius slightly higher than the desired periapsis (R_p) to begin aerobraking, to achieve a highly elliptic orbit with an eccentricity of 0.85. At apoapsis of the initial orbit (R_a), a second delta-v (delta-v2) of approximately 87 ft/sec, achieved through propulsion, is needed to lower the periapsis into the atmosphere. When apoapsis is degraded to the desired circular orbit (R_{po}), a third delta-v (delta-v3) of approximately 350 ft/sec will be used to raise periapsis out of the atmosphere and circularize to parking orbit.

Aerobraking presents problems which require the use of a computer to analyze. A program, AEROB, has been developed which should provide, among other things, an estimate of the time required for an aerobraking maneuver. The details of this program where documented in the PDR-1 report.

3.0 PARKING ORBIT OPTIMIZATION

To develop a criteria for selecting a parking orbit about Mars, a simple analysis was performed to relate the Barge's mass to a parking orbit. It was assumed the parking orbit will be equatorial, and initially, it was assumed that the parking orbit will be circular. After the initial phase of the analysis, elliptical parking orbits were included in the analysis. Although operational considerations are important to the selection of the parking orbit, the analysis only evaluated the merits of various parking orbits in relation to the Barge's total mass. From this stand point this study must be considered incomplete. Logic dictates that there must be a tradeoff between operational requirements and the desire to minimize the mass of the Barge.

In general the analysis was able to conclude that elliptical parking orbits are prefered over circular ones, and that the perigee of the parking orbit should be as low as possible. Based on the analysis performed to date, a elliptical parking orbit with a perigee of 270 n.mi. and a apogee at the orbital altitude of Phobos is recommended.

The discussion below describes a procedure used to minimize the total fuel requirement by choosing an optimal parking orbit altitude (Alt_{po}). Because the barge carries all the fuel and equipment needed for the entire mission, the fuel requirements for each phase of the mission can be taken into account by considering only the total weight of the barge. The discussion will initially describe the analysis assuming circular parking orbits, and will than be modified later to account for elliptical parking orbits.

The total weight of the barge prior to capture (M_{total}) is comprised of the weight of the ascent and descent operations equipment, the weight of the fuel used to capture the Barge about Mars, and the weight of the empty barge. In equation form,

$$M_{total} = M_{empty\ barge} + M_{capture\ fuel} + M_{ascent\ operations} + M_{descent\ operations}$$

where: $M_{empty\ barge} = M_B$ = **constant** = mass of barge without capture fuel or A/D vehicles.

$M_{capture\ fuel} = M_{CF} = \mathbf{f(Alt_{po})}$ = mass of the fuel used to capture Mtotal around

Mars.

$M_{ascent\ operations} = M_{AV} = f(Alt_{po})$ = total mass of ascent vehicles and fuel.

$M_{descent\ operations} = M_{DV} = f(Alt_{po})$ = total mass of descent vehicles, payload, and fuel.

The functional dependencies, $f(Alt_{po})$, are not meant to be comprehensive, but are meant to show the dependence of the Barge's mass, M_{total}, on the parking orbit altitude. The vehicle performance required of the Barge to capture itself and its payload about Mars is a function of the parking orbit's altitude; likewise , the performance required of the ascent vehicles and the descent vehicles to go to and from the parking orbit is also a function of the parking orbit's altitude. In our analysis this performance is related to mass carried by the Barge. The assumptions and the equations used to relate this performance to Barge mass are outlined below.

- The delta-v required of a ascent vehicle is estimated in the following manner:

$$DV_{ascent} = DV_1 + DV_2 + DV_3$$

The first delta-v, DV_1, takes the vehicle from a surface velocity to a circular orbital velocity at the surface. A fudge factor of 1.2 is used to account for losses such as atmospheric drag and gravity drag.

$$DV_1 = Sqrt(Mu/R_{mars}) - Omega_{mars}R_{mars}$$

where: Mu is the gravitational parameter of Mars.
R_{mars} is the mean radius of Mars.
$Omega_{mars}$ is the rotational velocity of Mars.

The second delta-v sends the ascent vehicle from its circular orbit at the surface into a transfer orbit to the parking orbit.

$$DV_2 = Sqrt(Mu(2/R_{mars} - 1/A_t) - Sqrt(Mu/R_{mars})$$

where: $A_t = (R_{mars} + R_{po})/2$

The third delta-v serves to circularize the transfer orbit at the parking orbit.

$DV_3 = Sqrt(Mu/R_{po}) - Sqrt(Mu(2/R_{po} - 1/A_t)$

where: R_{po} is the radius of the parking orbit.

The mass of the fuel required to achieve this performance is estimated using the ideal rocket equation.

$$M_{ascent\ vehicle} = M_{dry\ ascent\ vehicle} exp(DV_{ascent}/(I_{sp}G_0))$$

$$M_{ascent\ fuel} = M_{ascent\ vehicle} - M_{dry\ ascent\ vehicle}$$

where: I_{sp} = 260 sec. for the ascent vehicle.

$M_{ascent\ vehicle}$ is the mass of a fully fueled ascent vehicle.

$M_{dry\ ascent\ vehicle}$ is the mass of a ascent vehicle without fuel.

G_0 is the acceleration constant.

According to the mission senerio outlined in the Martian Operations section of this report, the barge will carry two ascent vehicles and enough fuel for three ascent missions. The mass of all the ascent vehicles and their fuel, $M_{ascent\ operations}$, is computed in the following manner:

$$M_{ascent\ operations} = 2M_{dry\ ascent\ vehicle} + 3M_{ascent\ fuel}$$

• The delta-v required of a descent vehicle, $DV_{descent}$, is approximated by a Hohmann transfer from the parking orbit to the surface. It is assumed that the atmosphere provides all of the propulsive braking needed for a soft landing on the surface.

$$DV_{descent} = Sqrt(Mu/R_{po}) - Sqrt(Mu(2/R_{po} - 1/A_t))$$

Again, as outlined in the Martian Operations section, the mission senerio calls for three descent vehicles. Each descent vehicle is to have a different payload and each vehicle will have to carry a fully fueled ascent vehicle. The mass of the three descent vehicles are computed as follows:

$$M_{descent\ vehicle\ \#1} = M_{dry\ descent\ vehicle} + M_{rover} + M_{ascent\ vehicle}$$

$M_{descent\ vehicle\ \#2} = M_{dry\ descent\ vehicle} + M_{lab/hab} + M_{ascent\ vehicle}$

$M_{descent\ vehicle\ \#3} = M_{dry\ descent\ vehicle} + M_{ascent\ vehicle}$

where: M_{rover} is the mass of the rover vehicle.
$M_{dry\ descent\ vehicle}$ is the mass of a dry descent vehicle.
$M_{lab/hab}$ is the mass of the Lab/Hab.

Using the ideal rocket equation, the mass of the fully loaded descent vehicles are computed in the following manner:

$M_{descent\ \#1} = M_{descent\ vehicle\ \#1} \exp(DV_{descent}/(I_{sp}G_o))$

$M_{descent\ \#2} = M_{descent\ vehicle\ \#2} \exp(DV_{descent}/(I_{sp}G_o))$

$M_{descent\ \#3} = M_{descent\ vehicle\ \#3} \exp(DV_{descent}/(I_{sp}G_o))$

where: I_{sp} =260 sec. for the descent vehicles.
$M_{descent\ \#(1,2,3)}$ are the masses of the fully loaded descent vehicles.

The total mass of the descent vehicles and the required fuel to be carried on the Barge, $M_{descent\ operations}$, is computed in the following manner:

$M_{descent\ operations} = M_{descent\ \#1} + M_{descent\ \#2} + M_{descent\ \#3}$

- The delta-v ($DV_{capture}$) required to capture the Barge about Mars, assuming a single propulsive manuver, is computed by:

$DV_{capture} = Sqrt(V_{approach}^2 + 2Mu/R_{po}) - Sqrt(Mu/R_{po})$

where: $V_{approach}$ is the hyperbolic excess velocity approaching Mars.

Again using the ideal rocket equation, the total mass of the Barge before the capture manuver is obtained with:

$M_{total} = M_f \exp(DV_{capture}/(I_{sp}G_o))$

where: M_f is the mass of the Barge after the capture manuver.

$M_f = M_{empty\ barge} + M_{ascent\ operations} + M_{descent\ operations}$

$I_{sp} = 450$ sec. for the capture manuver.

Table 3.1 contains the value of the parameters used in the analysis.

Figures 3.1 & 3.2 show that the delta-v requirements for ascent and descent increase with parking orbit, as would be expected. As a result, the mass of fuel also increases as shown by the behavior of the total weights for the ascent vehicle (Fig. 3.4) and the descent vehicle (Fig. 3.5). Fig. 3.3 shows that the capture delta-v requirement decreases as parking orbit increases, but Fig. 3.6 shows that the capture fuel mass increases rather than decreases for higher parking orbits. Even though the delta-v decreases, the fuel requirement increases. The reason for this result is that the higher parking orbit causes the total mass of the barge to increase because of the additional fuel required for A/D operations.

Figures 3.4-6 demonstrate that each mass componet of M_{total} increases as parking orbit altitude increases. Therefore, the total mass of the barge will also have to increase for higher parking orbits. This result is displayed in Fig. 3.7. So then, to minimize the total mass requirement for this particular scenario, the lowest possible parking orbit should be chosen. Choosing 270 nm. as the parking orbit altitude, the total mass of the barge would have to be approximately 700,000 lb.

At this point, the parking orbit optimization procedure has been demonstrated. Other scenarios can be defined and their parking orbits optimized by modifying the optimization procedure for the new scenario. One possibility is to capture into an elliptic orbit rather than a circular orbit. Applying the optimization procedure to the elliptic orbit scenario requires some minor modification of the delta v calculations.

$$DV_{capture} = Sqrt(V_{approach}^2 + 2Mu/R_{po\_p}) - Sqrt(Mu(2/R_{po\_p} - 1/A_{po})$$

where: R_{po_p} is the radius of the parking orbit at perigee.

$A_{po} = (R_{po\_a} + R_{po\_p})/2$

R_{po_a} is the radius of the parking orbit at apogee.

$$DV_{descent} = Sqrt(Mu(2/R_{po\_a} - 1/A_{po})) - Sqrt(Mu(2/R_{po\_a} - 1/A_{te}))$$

where: $A_{te} = (R_{po\_a} + R_{mars})/2$

$$DV_2 = Sqrt(Mu(2/R_{mars} - 1/A_{te}) - Sqrt(Mu/R_{mars})$$

$$DV_3 = Sqrt(Mu(2/R_{po\_a} - 1/A_{po})) - Sqrt(Mu(2/R_{po\_a} - 1/A_{te}))$$

Another major difference is that two optimization parameters must be determined; the altitude of the apoapsis and the altitude of the periapsis.

Making those changes for the elliptic parking orbit scenario, the parking orbit optimization is much more effective in reducing the total mass of the barge. Using an arbitrary apoapsis, the plot in Fig. 3.8 for total barge mass vs. parking orbit shows that the optimal periapsis will be as low as possible. To optimize the apoapsis of the parking orbit, a periapsis is chosen (270 nm.). In Fig. 3.9, a plot of total barge weight vs. parking orbit apoapsis shows that the higher the apoapsis is the better. To provide the greatest mission flexibility, an apoapsis at the orbit of Phobos is chosen. An elliptic orbit with apoapsis at Phobos yeilds a total barge mass savings of 70,000 lb. over the 270 nm. circular orbit.

4.0 MARTIAN OPERATIONS

4.1 SCENARIO

A Martian surface operations mission must serve two purposes. The mission should maximize scientific knowledge gained and serve as a precursor to a permanent manned presence on Mars. The baseline scenario chosen for the surface operations mission involves the manned exploration of two sites and the area between them. The major components used in this scenario are:

3 Descent stages (1 as a backup)
2 Ascent stages (1 as a backup)
Laboratory/Habitat
Shirt-sleeve Rover

The scenario begins with the propulsive landing of the first ascent/descent package at Site 1. This package includes the shirt-sleeve Rover and carries to the surface a crew of four. The crew then separates into two pairs. The first pair remains in the vicinity of the landing site and begins scientific exploration/experimentation. After an allotted time period, they return to the orbiting vehicle. The second pair departs from the landing site in the shirt-sleeve Rover and begin the journey to Site 2, conducting scientific exploration/experimentation along the way. Upon their arrival at Site 2, the second ascent/ descent package descends to Site 2. This package consists of the ascent vehicle of the first ascent/descent package, a new descent stage, and the Lab/Hab. This second package also carries to the surface a crew of two. At Site 2, the four merge in the Lab/Hab and continue scientific exploration/experimentation until the termination of the surface operations.

Assumptions:

1. Enough power can be generated by the Rover to span the 450 km distance between Site 1 and Site 2

2. The Rover will have the mobility to travel between Site 1 and Site 2

3. Experimentation and exploration will be conducted during the Rover's journey

4. The Rover will furnish life support for a crew of two for a minimum of thirty days

5. All ground crew will be able to protect themselves in the event of a solar flare

A detailed schedule of the surface operations mission follows.

Day

1-5 › First ascent/descent package descends to Site 1 (crew of 4)
Ascent/Descent Vehicle separation and landing at Site 1
A/D Vehicle (propulsive landing)
Rover (controlled impact)
Systems check
Communications established
Rover deployed
Rover equipment verified through preliminary data acquisition
Crew of two depart in Rover

5-17 › Ascent/Descent Vehicle
Continued exploration/experimentation
Launch preparation
Rendezvous with orbiting vehicle

18-25 › Exploration of Moons

5-25 › Rover
Continued exploration/experimentation during journey to Site 2
12 days mobile (assuming travel at 5 km/hr, 8 hr/day)
8 days stationary

25 › Rover established at Site 2
Second ascent/descent package descends (crew of 2)
Ascent/Descent Vehicle separation and landing at Site 2
A/D Vehicle (propulsive landing)
Lab/Hab (controlled impact)
Ascent/Descent Vehicle systems check; communications established with orbiting vehicle and Rover
Rover and A/D crew now inhabit the Lab/Hab

26-56 › Continued exploration/experimentation

57-60 › Long term storage of Rover and Lab/Hab equipment/systems

60 › Ascent of crew of four and rendezvous with orbiting vehicle

4.2 SITE SELECTION

In developing landing site criteria, three primary areas of investigation must be considered: biology, geology and meteorolgy. The goals of the mission must be analyzed before the landing sites can be selected. Is the purpose of the mission to find life forms, study the geology of the planet further, study the atmosphere, map the planet, beat the Russians there or a combination of the above? Also, the terrain of the landing site must be studied before landing, to determine if it is feasible to land at that particular site.

Finding life forms is one of the primary goals of the mission. According to Joshua Lederberg, the most suitable place for life forms to exist is at a latitude of plus or minus 65 degrees or higher. The probability of biological forms existing as we know them is much greater at this latitude, where the polar caps have receded. Of course, there may be other types of biological beings that may find the equatorial latitudes more suitable.

A ground rule for this mission is that the manned landing will be made within plus or minus 5 degrees of the equator. The polar areas will not be considered. Although this may sound restrictive, the geology of the equatorial region of Mars varies greatly.

Three candidate sites for landing have been chosen, Biblis Patera (3 degrees north, 124 degrees west), Ulysses Patera (3 degrees north, 121 degrees west), and Pavonis Mons (approximately 0 degrees, 110 degrees west). Figure 4.2.1 shows the location of these landing sites. These volcanoes are surrounded by the volcanic plains of Tharsis, located in the Tharsis Montes region. This is a region of relatively young lava flow that emanated from the Tharsis volcanoes. They consist mostly of shield volcanoes varying in age from relatively young to intermediate to ancient. Volcanic sites were selected to satisfy the requirements of the University of Wisconsin study. They wish to land near a volcanic site to build a Mars base. The Martian soil will be used to cover the base to protect it from solar radiation.

Another candidate site is the Schiaparelli crater (approximately 5 degrees south, 345 degrees west). This large crater is surrounded by several different types of terrain. The varied terrains of the Schiaparelli crater consist of ridged plains, heavily cratered uplands, and mountainous terrain. The ridged plains are relatively old lava plains showing prominant wrinkle ridges and other features similar to the Luna Maria. The heavily cratered uplands consist of hilly and cratered terrain. In many areas, the

intercratered regions are relatively smooth and dissected by small, dry valleys. The mountainous terrain is ejecta and uplifted blocks of ancient terrain caused by impacts. The ridged plains surrounding the crater would be the best place to land in this area.

The recommended sites are two of the volcanoes, Pavonis Mons and Biblis Patera. The distance between these two sites is approximately 450 km. This distance is within the range of the Rover's exploration capabilities. In the current scenario, both sites and the area between them will be explored by the astronauts. These sites must be surveyed in greater detail upon arrival at Mars (both of the original landing sites for the Viking landers were changed because of unsuitable terrain). The final site selection will be determined upon arrival. The two remaining sites, Ulysses Patera and the Schiaparelli Crater, should be considered alternate sites. The terrain of Ulysses Patera is similar to that of Biblis Patera and Pavonis Mons.

4.3 ASCENT / DESCENT VEHICLE

The Ascent / Descent (A/D) Vehicle will provide the transportation for the crew to and from the surface of Mars. A total of three ascent sections and two descent sections will be sent to Mars, including one descent and one ascent section held in reserve for emergencies.

4.3.1 Operational Schedule

The ascent and descent of the A/D vehicle will require an operational plan to insure that no aspect of the mission will be overlooked. The following schedules are an attempt to illustrate the general operational schedule.

4.3.1.1 Operational Schedule: Descent

The following section describes the step-by-step procedure for descent to the Martian surface.

1. Crew enters the Ascent/Descent (A/D) Vehicle, performs system checkout procedures.

2. A/D vehicle separates from orbiter, makes a deorbit burn, and then begins free fall descent from 270 nmi. orbit.

3. When the A/D vehicle reaches an altitude of about 100,000 ft., parachutes are deployed to reduce speed.

4. After the parachutes have reduced the velocity to 1500 ft/sec, the heat shield is jettisoned pyrotechnically, with solid rocket boosters being used to reduce the velocity further, to 440 ft/sec. At this point, the liquid fuel descent engine is activated.

5. Parachutes are jettisoned after powered descent begins.

6. Landing gear is extended within 1000 ft of the planet surface.

7. As the A/D vehicle makes its final approach, the pilot takes over from the automatic navigation and control computer to guide the last stages of the descent.

8. A/D vehicle touches down on the planet surface, with automatic cutoff of engine upon touchdown.

9. Safety system checks are made and landing gear is adjusted hydraulically to provide a level working environment.

4.3.1.2 Operational Schedule: Ascent

The following section describes the step-by-step procedure for the ascent from the planet surface.

1. Crew enters A/D vehicle, performs system checkout procedures, detaches any links between the ascent and descent portions of the vehicle.

2. Ascent vehicle begins ascent with solid rocket boosters for initial delta-V boost.

3. Upon burnout, the solid rocket boosters are jettisoned and the liquid fuel rocket engine is activated for insertion into rendezvous orbit.

4. Upon reaching rendezvous orbit, the ascent vehicle will maneuver and dock with the orbiter. When docking is complete, the vehicle systems will be powered down, and the crew will transfer to the orbiter.

5. After a safety inspection has been made, the ascent vehicle will be joined with a new descent section, refuelled and resupplied, and prepared for the second trip down to the planet surface.

4.3.2_Systems

This section describes some of the systems onboard the Ascent/Descent Vehicle.

SCIENCE EXPERIMENTS

In the exploration and study of Mars, a broad spectrum of equipment will be needed to conduct science experiments and other activities. The equipment used on the A/D vehicle breaks down into two areas: analytical hardware and support hardware. One of the primary areas of study will be the geological makeup of Mars. The properties of the Martian soil will be studied to determine its usefulness as raw material for present and future material needs. To further these objectives, microscopes,cameras and spectrum analyzers will be used to examine samples. A mass spectrometer, refractometer and X-ray diffractometer will be used to study various rocks and their structure. A network of unmanned probes will be used to monitor seismic activity over a large area of the planet surface.

To obtain the samples needed for study in the lab, support equipment will be used. These items will include excavation equipment, remote manipulators, ovens, storage facilities, and core sampling equipment. From these efforts, samples will be obtained which will include examples of sedimentary patterns, core samples, different types of surface soil, water samples, and so on. An estimate of weight, volume and power requirements are listed below. These numbers were taken from a study done by the Boeing Company on the subject of science payloads and weights.

Description	Weight(lb)	Power(watts)	Volume(ft**3)
Analytical hardware	794	264	15.9
Support hardware	573	1603	43.5
Samples & Data Return	912	150	9.2
Totals	2279	2017	68.6

ONBOARD POWER SUBSYSTEM

This subsystem supplies, regulates, and distributes all electrical power used in the vehicle. On current manned space systems, onboard power is supplied by fuel cells. In this process, gaseous hydrogen and oxygen are chemically combined to produce electric power, with waste heat and water as by-products. The water can be purified and used by the crew for drinking and bathing.

Another option which may be of greater value is that of radioisotope thermoelectric generators (RTG's). These small nuclear power units can provide anywhere from 10KW to 100KW, in a very small space. Due to the length of this mission, it is proposed that RTG's be the main source of power for the A/D vehicle, with fuel cells being retained as a backup.

PYROTECHNICS

Pyrotechnics are explosive devices which are used to separate or deploy systems on the A/D vehicle. Pyrotechnics will be used to release the parachutes. After the parachutes have slowed the descent sufficiently, the heat shield will be dropped pyrotechnically, leaving the descent rocket engine exposed and ready to begin the powered descent portion of the A/D vehicle entry. As the vehicle approaches the ground, pyrotechnics will be used to deploy the landing gear to its operational position. During the ascent portion of the mission, pyrotechnics will probably be used to separate the ascent stage from the descent stage.

HEAT SHIELD

The heat shield provides protection for the A/D vehicle as it begins its descent into the Martian atmosphere. Its shape will be similar to a hemisphere, but it will extend upward in a cylindrical form to protect the sides of the vehicle. The shield will utilize both radiative and ablative cooling, using ablative structures for very high temperature regions, and radiative cooling for areas of lesser heating. After the parachutes have slowed the vehicle to 440-500 ft/sec, the heat shield will be jettisoned pyrotechnically.

NAVIGATION & GUIDANCE SUBSYSTEM

The navigation and guidance subsystem provides the equipment for measuring the position and velocity of the A/D vehicle, as well as the data processing capabilities for generating the control functions for stabilization and control, accomplishing the descent and ascent, and orbital rendezvous. Equipment in this section includes landing radar, rendezvous radar and

transponder, an alignment optical telescope, inertial measurement unit, guidance computers, power and servo assembly and display units.

Using the landing radar, the altitude and velocity can be measured. This external data is then compared to inertially derived data for a double check against possible errors. The rendezvous radar is used to track the orbiter as it passes overhead, providing information on orbiter position and range.

LIFE SUPPORT

In the area of environmental control, the primary areas of concern for the A/D vehicle are atmosphere revitalization, cabin pressure, heat transport and equipment cooling, and water management. Atmosphere revitalization provides oxygen to cool and ventilate the crew quarters, controlling recirculation and temperature of cabin air. The cabin pressure system maintains the correct amount of oxygen, preventing overpressurization. The heat transport section is a closed loop system that use a coolant circulating through heat exchangers to remove waste heat from cabin air, and uses the heat to warm other portions of the vehicle. Equipment cooling is accomplished by passing coolant through plates which are attached to the equipment, serving as a heat sink. The water management system is primarily concerned with removal of excess moisture from cabin air, production of potable water from the fuel cells, and use of water as a coolant.

COMMUNICATIONS SUBSYSTEM

This subsystem contains a signal processing section and two major radio frequency(RF) links, one operating out of the VHF range, the other out of the UHF range, or S- band. The signal processing section controls the various communications signals during the different operating modes. The RF links provide two-way voice communications between the A/D vehicle and the orbiter, as well as between the Rover or Lab/Hab and the A/D vehicle. Information to be processed includes voice, video, telemetry, tracking and biomedical data.

PROPULSION SUBSYSTEM

The descent propulsion stage provides the thrust level and velocity change necessary for controlled descent to the planet surface after the separation of the heat shield. It is assumed that the engine will use liquid fuel, pressure fed with a bi-propellant mixture. The descent engine will be

throttleable and gimballed to enable the pilot to guide the descent to the fullest extent possible.

The propellant will be supplied from baffled tanks pressurized by helium. Redundancy will be built into the propellant feed and pressurization process to ensure that engine operation will not be interrupted by one component.

The descent propulsion stage will be fully contained within the descent structure, using a single throttleable engine. The gimbal mounts should provide motion within a 12 degree solid angle, giving control about the pitch and yaw axes during descent. The fuel and oxidizer will be stored in several pairs of tanks, each of which will be fed into a common engine feed line.

The propulsion requirements for the ascent vehicle are Mars lift off, ascent and rendezvous with the orbiter. The propulsion system consists of liquid propellant rocket engines with their own propellant feed and pressurization sections, and strap-on solid rocket boosters.

The ascent propulsion stage is wholly contained within the ascent structure. The solid rocket boosters are utilized to provide the major thrust required for lift-off. Once the rocket boosters have burned out, they will be jettisoned. The ascent stage will employ throttleable engines (liquid) to develop the remaining thrust that is needed. The engine will be gimballed to provide trim control of pitch and yaw during ascent. The propulsion feed and and pressurization section will be redundant for additional reliability.

IN-SITU PROPELLANT PRODUCTION

In-Situ Propellant Production (ISPP) will be carried out as an experiment on this mission, in the hope that it will serve as a useful tool for reducing the weight of fuel carried on future space flights. With ISPP, the Martian atmosphere and surface materials are used as raw materials to make fuel and oxidizer for use as rocket propellant and rover fuel. The Martian atmosphere contains an abundance of carbon dioxide, which can be heated to produce carbon monoxide and oxygen. The oxygen can be separated out and stored for use as an oxidizer. Fuel production is more difficult as the hydrogen atoms necessary for LH_2 production are locked up in permafrost ice in the Martian soil or in water ice at the polar ice caps. Mining the permafrost would be a formidable task, while an expedition to the polar ice caps would be outside the range of this mission. For these reasons, it is recommended that methane be brought to Mars to serve as the fuel for this experiment.

LANDING GEAR

The criteria for landing gear operation shall be that the gear can withstand speeds of up to 10 ft/sec vertical velocity and 5 ft/sec horizontal velocity. The configuration of the landing gear will be similar to that of the Lunar Excursion Module. The gear will be of cantilever type, mounted externally, with four legs at equal spacing. The components consist of a primary strut, a footpad, secondary struts, driveout and downlock mechanisms and additional support frames. The gear will be retracted until the heat shield has been jettisoned. At that point, the locks will be pyrotechnically released, with the stored energy being used to drive out the gear into its operational position, where the down locks will secure it in place. Once the A/D vehicle has landed, a hydraulic system will be used to make any changes necessary to level the vehicle .

REACTION CONTROL SUBSYSTEM

This subsystem provides low level translational impulses that provide control of the vehicle during the descent, as well as rendezvous and docking during ascent. The configuration should be 20 thrust chambers, in clusters of five around the sides of the vehicle. For each cluster of chambers, two nozzles should be oriented in a vertical but opposite position relative to the vehicle, with another pair normal to the first two.The last nozzle is used for translational motion. By using any pair of chambers, the attitude about the pitch,roll and yaw axes can be maintained. The thrust is provided by the reaction of N_2-H_4 and N_2-O_4.

The RCS can be administered as a fully automatic, semi-automatic or manually operated system. During the early descent stages, the preprogrammed flight instructions control the ship. In the final approach, the pilot may elect to choose semi-automatic or manual control to insure a safe arrival.

4.3.3 Performance

4.3.3.1 Descent Vehicle Performance

All mission scenarios, regardless of complexity, demand that fuel requirements be minimized. For this reason, a scenario was developed with the assumption that a descent from parking orbit (h = 270 n.mi.) to Mars surface could be accomplished with two propulsive burns: one for deorbit, and a second one at an altitude of approximately 10,000 ft.

In this scenario the descent vehicle uses a deorbit burn to descend from parking orbit along an elliptical orbit of h_A = 270 n.mi. and h_p = 1 n.mi. As the vehicle enters the atmosphere, a heat shield and other drag devices are employed to reduce velocity to subsonic level. Parachutes are then deployed to continue the braking at subsonic speeds. Between 10,000 ft. and 5000 ft. altitude the heat shield is jettisoned and the Lab/Hab and A/D vehicle are separated. The Lab/Hab retains the parachutes and descends to a controlled landing; any remaining velocity is absorbed upon impact by a honeycomb-composite controlled landing system. The A/D vehicle decelerates to a soft landing using a variable thrust propulsive system

It was decided to separate the Lab/Hab and the A/D vehicle because of possible structural damage that the Lab/Hab might sustain during the ascent portion of the mission. Also, it was decided that separation could be carried out safely only at subsonic speed. Consequently, separation is delayed until an altitude of approximately 10,000 ft.--to allow aerodynamic drag to slow the vehicle as much as possible.

In order to analyze this scenario, a program (TKRUS) was written to simulate the descent of a spacecraft through the Martian atmosphere. The equations of motion for flight over a spherical (non-flat) world were integrated from an altitude of 126 n.mi. (the semi-latus rectum of the ellipse mentioned earlier; see Figure 4.3.3.1.1) to the surface. A variable-order integration routine was supplied by the U.T. Aerospace Engineering faculty to integrate the equations of motion. The Martian density was calculated from a two layer "Standard Atmosphere" model supplied by Lincom Corp. of Houston. Atmospheric temperature was calculated only for altitudes below 16.5 n.mi. so that a straight line approximation could be used; the temperature was used in calculating Mach numbers. Wave drag and wake drag were not included in the computations, and induced drag was assumed zero.

The semi-latus rectum of the ellipse mentioned earlier (see Point A of Figure 4.3.3.1.1) was used as the initial altitude for TKRUS. Initial velocity and flight path angle were determined as a function of the orbital ellipse and the altitude. The values are: h_0 = 126 n.mi.; V_0 = 10,563 ft/sec with respect to the surface; and initial flight path angle = -4.0^0. Baseline values for the descent vehicle were estimated at: m = 115,000 lb.; A = 1,000 ft^2; and C_d = 1.4. TKRUS was run using these values; then TKRUS was run to see how the results varied for different values of mass, area, and C_d. Mass was varied from 50,000 lb. to 150,000 lbs.; area was varied between 1,000 ft^2 and 10,000 ft^2; and C_d was varied from 1.0 to 2.0. Results of these runs have been plotted in Figures 4.3.3.1.2 - 5.

Several conclusions were drawn from the results of the computer runs:

1. For altitudes above 25.8 n.mi., the spacecraft trajectory is independent of mass, area, or C_d (see Figure 4.3.3.1.6);
2. Maximum velocity (V_{max} = 11,090 ft/sec) is attained at 31.3 n.mi.;
3. With m = 115,000 lbs.(baseline value), subsonic velocity cannot be reached;

In short, the scenario described above will not work using the estimated baseline figures. No supersonic drag devices could be found or envisioned which would slow the baseline craft to subsonic speed.

The initial scenario was abandoned, and a new scenario using chemical propulsion was developed. Between deorbit and 16 n.mi., the old and new scenarios are identical; but at 16 n.mi. , deceleration is aided by the deployment of three supersonic parachutes. At an altitude of 10,000 ft., two solid rocket boosters (SRBs)--attached to either side of the descent vehicle--are ignited to obtain a Δv of 1220 ft/sec. At some altitude above 5000 ft the depleted SRBs and the heat shield are jettisoned, and the descent vehicle is then separated into its two components, the Lab/Hab and the A/D vehicle. Between 5000 ft. and touchdown, both vehicles continue to decelerate using variable thrust engines.

The rationale for placing the parachute deployment altitude at 16 n.mi. is as follows:

1. Atmospheric effects can be ignored above an altitude of 25.8 n.mi., so the parachutes would be deployed at some lesser altitude;

2. The parachutes should be deployed as soon as possible to maximize deceleration due to drag;
3. Parachutes deployed near 25.8 n.mi, where density is relatively small, might not open;
4. An altitude of 16 n.mi. was chosen as a reasonable compromise between minimum fuel requirements and safety requirements.

The separation scenario was run through TKRUS to verify feasibility. The initial mass value was increased to 150,000 lbs. to account for the propulsion systems for the descent vehicle, the Lab/Hab, and the A/D vehicle. All other baseline values remain unchanged. Results have been plotted in Figures4.3.3.1.7 and 4.3.3.1.8.

Below is a summary of important events in the descent scenario, along with some pertinent data:

EVENT	h(ft)	V(fps)	m(lb)	A(ft2)
Start of atmospheric drag effects	157,000	11,000	150,000	1,000
Deployment of supersonic parachutes (1000 ft2 ea.)	95,600	10,300	150,000	3,000
SRBs ignite	9,910	1,660	150,000	3,000
End of propulsive burn, heat shield and SRBs jettisoned, Lab/Hab and A/D Vehicle separated	5,000	440	130,000	3,000

Table 4.3.3.1.1 Descent Scenario

4.3.3.2 A/D Vehicle Performance

During the first portion of the descent trajectory, the vehicle has made the descent as a single vehicle. After the solid rocket boosters have burned out, the A/D vehicle is separated from the Lab/Hab. The A/D vehicle then begins to make a controlled descent using a liquid fuel engine.

For the purposes of this analysis, the descent engines were to be activated at 5000 ft., with a velocity of 440 ft/sec. The thrust of the engines was assumed to be variable, so a thrust control loop was introduced into the equation. The equations of motion for the A/D vehicle were integrated, with the objective of optimizing the thrust control to use the least amount of fuel during the descent. The guideline for this minimization was a parabolic projection of velocity vs. altitude, with (440,5000) providing the high boundary value, and (5,0) serving as the low boundary value (see Fig. 4.3.3.2.1). Using the starting conditions mentioned above, and using a value of C_d = 1.4, I_{sp} = 260 sec, and flight path angle = 45°, the end result was an optimized burn of 8480 lb. of fuel over a 28 second period.

4.3.3.3 Ascent Vehicle Performance

The vehicle which will return the astronauts, Martian soil and rock samples, and any other payload items from the surface of Mars to the main vehicle in orbit is termed the Ascent Vehicle. The overall design of this vehicle is highly dependent on the mission requirements, such as the stay time on the surface, whether or not the astronauts will be living in the ascent portion of the combined ascent/descent vehicle, and most importantly, the altitude and inclination of the main vehicle's parking orbit about Mars.

The characteristics of the ascent trajectory which minimize drag losses maximize gravity losses, and those which minimize gravity losses maximize drag losses. To reduce drag losses it is necessary to get into space as soon as possible. This indicates a vertical launch. This, however, increases gravity losses since the total weight is acting against the thrust. To reduce gravity losses, a low flight path angle is desired as only W sin(ø) acts opposite the thrust, (where ø is the angle between the horizontal and the flight path). A lower angle will increase drag losses as the vehicle stays in the atmosphere for a longer period of time. As a rough estimate of the velocity losses (including the negative loss, or gain, due to the planet rotation), the following calculations were performed:

Desired circular orbit velocity: $V_c = (mu/r_c)^{1/2} = 10{,}896.9$ ft/sec
(r_c = radius of Mars + 270 n. mi.)

Add estimate of losses due to atmosphere and gravity:

$V_{loss} = .2 \times V_c = 2179.4$ ft/sec

Subtract eastward velocity for an equatorial launch site:

$V_{rot} = r_{Mars} \times w_{Mars} = 791.2$ ft/sec

The total delta V for an equatorial, 270 n. mi. altitude ascent:

$V_{tot} = V_c + V_{loss} - V_{rot} = 12285$ ft/sec

This estimate assumes a conservative 20% drag and gravity loss, and neglects nozzle losses due to overexpansion of the exhaust in the lower atmosphere. The actual specific impulse attained will increase from a minimum at the surface to a maximum in vacuum as the vehicle climbs out of the atmosphere. The theoretical maximum specific impulse is attained when the nozzle is infinitely long, and exhaust pressure is equal to zero. For the specific impulses

used in computer simulations, a constant value halfway between earth sea level and vacuum values was assumed. The ideal velocity of the vehicle at each point on the trajectory is not a function of the thrust, but only of the total impulse available. The thrust to weight ratio must be greater than one for the vehicle to leave the surface, however. The desired result of the ascent vehicle performance study, then, is given the required burnout conditions of velocity, altitude, and flight path angle, find the trajectory that results in the lowest engine and propellant weight.

In order to determine the required delta v and the ratio of propellant mass to total mass to reach the parking orbit, a numerical integration of the ascent trajectory must be used. Two separate programs were written, but neither program gave the desired results. A FORTRAN numerical integration program that takes the effects of gravity (two body only), atmospheric drag, rotation of the planet, and thrust into consideration was written. The program did not work properly, however, and could not be made to work in the available time.

Originally, the trajectory program was to have used a flat planet model, with no thrust vector or flight path angle control. A subroutine was written to compute the density as a function of the altitude, to determine the drag force on the vehicle. The only controlled variables were the thrust level and the flight path angle at launch. The problem with this simplified approach is that the trajectory will always describe a parabola with an unpredictable apoapsis. Since the purpose of the program was to integrate a trajectory from the surface (at a launch site on the Martian equator) to a circular, near-equatorial inclination parking orbit of specified altitude, many runs of this program must be made to find the right combination of thrust and initial flight path angle to achieve orbit, with no guarantees that the trajectory was optimum. For the reasons mentioned, this approach was not used. Instead, a more sophisticated program was written which uses specified boundary-condition orbit fitting to target the trajectory. This method allows a choice of any combination of three of the following four target parameters: altitude, downrange angle, velocity, and flight path angle. In addition, this method provides a guidance algorithm for spacecraft in powered flight. The method will produce a trajectory that is near-optimal for fuel consumption, which will affect the amount of fuel needed. For the Mars Ascent Vehicle, the target parameters are: altitude = nominal circular orbit altitude, velocity = circular orbit velocity, flight path angle at orbit insertion - zero. Choosing the flight path angle as zero at the orbital altitude uniquely determines the downrange or transfer angle. The method also provides for the determination of the near-optimal thrust level and the required thrust angle to reach the target point. A constraint was put on the thrust to limit the acceleration on the vehicle to 3 g's, or 96.5 ft/sec^2.

The program was written and it was found that the equation which computes the downrange angle appears to be in error.

The program described above was to have been used to determine the required delta V and mass ratios for a range of final orbit altitudes. The process was to be repeated for some specific liquid propellants with differing specific impulse, storage requirements, and feasibility for the Manned Mars Mission. On the basis of these results, the "ideal" propellant was to be chosen.

4.3.4 Propellant Selection

The liquid propellant to be used for the Ascent Vehicle will have to meet the following requirements: average to very good specific impulse in order to limit the amount of fuel needed, must be storable for a long period of time without decomposition or change of state, and the equipment required for storage must have reasonable power and space requirements. The requirements tend to be in conflict, since the most energetic propellants with the highest specific impulse also have the most demanding cryogenic storage requirements. The more energetic propellants bring about a condition of diminishing returns, since more storage tank insulation, heavier cryogenic equipment, and extra fuel to allow for losses must be carried. The Space Shuttle main engines use liquid oxygen for oxidizer and liquid hydrogen for fuel. Hydrogen has the most energy per unit mass of fuel available, but it has the disadvantages of having a low energy content per unit volume, it must be liquified at 32.9 K, is usually stored at 20 K, and will have about 1% losses during a 220 day mission when it is stored, even at the low temperatures specified. Hydrogen must be stored as a liquid to be practical as a fuel, since the volume of liquid hydrogen is 700 times less than an equal mass of hydrogen gas. Liquid oxygen is somewhat better, as it must be liquified at 154.8 K and stored at 90 K. Liquid oxygen also requires about eight times the weight of liquid hydrogen if they are to be used as the propellant combination. On the basis of these storage requirements, liquid hydrogen does not appear to be a viable fuel, while liquid oxygen may still be a viable oxidizer. The possibility of storing the hydrogen as methane (CH_4) with a boiling point of 109 K and a storage temperature of about 80 K, and then converting the methane back into pure hydrogen before burning it has been investigated. There is very little information available on the technology required to do this, but it is still being considered, since the liquid hydrogen-liquid oxygen propellant has a very good specific impulse of 360 seconds at earth sea level. The qualities of some fuel-oxidizer combinations are given below in Table 4.3.4.1. The table is by no means exhaustive, but it does give some indication of the choices available.

Fuel	Oxidizer	Mixture Ratio (by weight)	Specific Impulse (sec level), sec	Storability fuel	Storability oxidizer
Hydrogen	Oxygen	8	360	F-P	F-P
Hydrogen	Flourine	9.42	390	F-P	F-P
Hydrazine	$H_2 C_2$	1.5	245	G	G
Hydrazine	Oxygen	0.75	265	G	F-P
Hydrazine	Flourine	2	260	G	F-P

Storability : G = good; F = fair ; P = poor. "Good" storability means liquid can stored in ordinary tanks over long periods and at many temperatures without decomposition or change of state.

Table 4.3.4.1

On the basis of further research into the problem, it was determined that the most likely propellants for the Ascent Vehicle would need to be solid propellants, for the following reasons:

* Solid propellant rockets have no moving parts, no tanks, no injection system, and they require, as a rule, no cooling.
* They are easily stored, handled, serviced, and they are ready for launch at any moment.
* The lack of pumps, valves, and controls facilitates operation and minimizes failures. Reliability can be as high as 99%.
* The absence of turbomachinery and other accessories increases the obtainable mass ratio.

There are disadvantages in using solid propellants, however, such as a lower specific impulse than many liquid propellants (in the range of 250 to 300 seconds for solids), thrust control and termination is difficult, and the performance is quite sensitive to variations in temperature. In addition, the propellant must be stored at a sufficiently high temperature to avoid deterioration of the elastic properties, and increases in the ignition time. Since the solid rocket cannot be throttled, a liquid propellant rocket must be provided for the final insertion into orbit, and parallel staged with the solid propellant booster. To conform with the overall mission design of this study, it

has been decided that the ascent and descent vehicles will be sent to Mars on a precursor vehicle, which will remain in orbit about Mars. The propellants for the ascent and descent vehicles will have to be stored for long periods without excessive power requirements. The descent vehicle would most likely not have any solid propellant, since it's thrust will have to be finely controlled during the descent to the surface. Liquid monopropellants can be eliminated since all have poor specific impulse. A study of the available fuels and oxidizers suggests that the oxidizer most easily storable in space is hydrogen peroxide H_2O_2, which will not react with pure aluminum tanks, and has a freezing point of -0.4^0C. Since the temperature on the surface of Mars is approximately -59^0C, this oxidizer would have to be kept warm while the ascent vehicle is on the surface. There are no propellant storage requirements for the descent vehicle on the surface, since all of it's propellant will have been consumed during the descent. Two possible fuels that could be used with hydrogen peroxide are anhydrous hydrazine, N_2H_4; and unsymmetrical dimethylhydrazine (UDMH), $(CH_3)_2NNH_2$. UDMH has a freezing point of -52.0^0C, can be stored in mild steel, stainless steel, nickel, and aluminum tanks, and when combined with hydrogen peroxide has a specific impulse of 272 seconds. Hydrazine has slightly less desirable qualities, with a freezing point of 1.4^0C, good storability in aluminum tanks, and a specific impulse when combined with hydrogen peroxide of 265 seconds. Both UDMH and Hydrazine are hypergolic (ignite on contact) with hydrogen peroxide, which would simplify the rocket engine somewhat by making an igniter unnecessary. A combination of UDMH and Hydrazine was used on the Titan II ICBM, and the Ariane launch vehicle utilizes UDMH fuel and nitrogen tetroxide, N_2O_4, as oxidizer. The nitrogen tetroxide is used because it can be mixed with nitric acid and used in varying climatic conditions, but it is very toxic. For the Ariane's launch into low earth orbit, the propellants make up 90% of the initial mass.

The final selection of a propellant for the Ascent Vehicle will depend on the data generate ascent trajectory programs, the propellant storage requirements, and the constraints imposed by other phases of the mission; such as the Descent Vehicle (propellant compatibility for example), and the interplanetary vehicle's constraints on allowable payload mass.

4.3.5 Ascent Vehicle Staging

From the discussion above, it seems desirable to make the Ascent Vehicle a parallel staged rocket because of the storability advantages of solid propellants. The object this would be to have the solid propellant impart as much of the required delta V as possible, so as to reduce the amount of liquid propellant that has to be stored. A parallel staging of liquid propellant engines and solid propellant boosters which drop off after their fuel is spent may allow a lower mass ratio to be achieved. The liquid propellant would be retained, since the thrust must be throttled for at least a part of the trajectory, and solid rocket motors are not easily throttleable. Solid propellants do have the advantage of being relatively easy to store, as they require no cryogenic equipment, as discussed above. The Solid Rocket Boosters of the Space Shuttle use atomized aluminum powder for fuel and ammonium perchlorate for oxidizer, with some other compounds as catalysts and binders. An added advantage to using solid rocket boosters for later follow on missions to the same site is that the spent fuel casings could be recovered and either refurbished for another flight or used for some other purpose.

There are many solid propellants to choose from, but the driving criteria for selection would be the specific impulse (which implies a high release of chemical energy, and a low molecular weight of combustion products), the required temperature for stable long term storage, high density for a small chamber volume and weight, low temperature sensitivity over a wide range of storage and operating temperatures, ability to withstand repeated temperature cycling prior to operation without physical or chemical deterioration, and be chemically inert during storage and operation. In addition to the specific propellant, a suitable grain configuration to achieve the best thrust-time program must be chosen. Most likely, a configuration which yields a regressive thrust-time program would be chosen, since this would make it possible to keep the acceleration on the vehicle constant as propellant is burned and the total mass decreases. Since the Ascent Vehicle is likely to be small, the solid propellant with the highest density which meets the temperature and physical strength requirements would be chosen. The reason for this is that a high density propellant would keep the size of the solid rocket boosters down. The specific impulse should be as high as possible in order to keep the final mass ratio high.

4.4 ROVER VEHICLE

The Martian Rover has been designed to provide a shirt-sleeve environment for two persons on a twenty day journey over 280 miles (450 Km) of the Martian surface.

The Rover is estimated to weigh 11,000 lbs. A breakdown of this weight estimate is given below.

Power Source	4500 lb
Rover Structure	3000 lb
Expendables and Life Support System	2000 lb
Scientific and Navigation Equipment	1500 lb
Total Rover Weight	11000 lb

TABLE 4.4.1

A major subsystem of the Rover is the communications system. The purpose of the Rover's communication system is to transfer voice, video, and data to support the operation of the Rover and scientific experiments. The following guidelines were established for the Rover's communication network. The Rover must be able to communicate with the orbiting ship. The communications with the Orbiter include sending voice, video, experimental data, and experimental results from the Rover . It will not be able to communicate or send data directly to the Earth. The Rover must also be able to communicate with the astronauts when they are away from the Rover exploring the Martian surface. The main links of the Rover's communication system are shown in Figure 4.4.1. An antenna on top of the Rover will be used to aid in communications.

Data gathering and storage is a subsystem that is linked to the communications of the Rover. Scientific data will be obtained through the use of video cameras, scientific instruments, and sample collection. The surface samples gathered by the astronauts will be stored in the Rover and taken to the Orbiter in an ascent vehicle. Computers in the Rover will be used for data storage, data processing, performing calculations on the Rover's path of travel while traversing, monitoring any experiments which may be located on board the vehicle, and verifying the proper operation of all vehicle subsystems. However, the data storage on the Rover will be limited. Because of this, much of the data gathered and processed by the onboard computers will be sent to the orbiter for storage or relay to Earth.

The life support subsystems will support two astronauts for a minimum of 30 days. The environmental control system for the Rover will be open loop, therefore the Rover must carry all expendables. The airlock for the Rover will be a scaled down version of the airlock used in the Space Shuttle. The airlock of the Rover has a diameter of 48 inches and a height of 78 inches. The Shuttle's airlock is designed for use by two persons at the same time. The Rover's airlock will be used by one person at a time.

A major factor influencing the design of the Mars Rover is the surface environment. Two concerns are the temperature extremes and the Sun's radiation. Data from Viking I and II has shown that the temperatures of the surface of Mars can vary greatly from day to night and from sunlight to shade. The temperature extremes for the planet are -193 to 90 degrees Fahrenheit (-125 to 32 degrees Celsius). Because of these large temperature changes, the Rover body must be able to expand and contract to prevent any unwanted stresses on the structure. The Rover must also be heavily protected from the Sun's radiation since the Martian atmosphere does not provide adequate protection from solar radiation. The radiation shielding for the Rover consists of aluminum and water. The sides of the Rover are constructed of aluminum 1.5 inches thick. The radiation shielding for the roof will consist of the 0.5 inch aluminum roof plus a 3.0 inch layer of water between the roof and the inside ceiling of the Rover. Water is used in the roof because it has a good shielding per weight ratio and the roof is expected to experience the greatest amount of radiation. Radiation shielding adds considerable weight to the Rover. The aluminum structure weighs approximately 3000 pounds. The water for shielding will be generated by the fuel cells.

Viking I landed in a crater plain near the proposed landing sites for the Mars mission. Pictures from Viking I show that the Rover will have to travel over rocks ranging from pebble size to boulders several yards across. Therefore, the minimum ground clearance of the Rover will be 36 inches. The Rover has legs which can be manipulated to raise the Rover, giving a maximum clearance of 54 inches. The maximum operating power for the Rover will allow it to negotiate slopes of up to 30°. When the legs of the Rover are extended, the Rover's center of gravity will be high off the ground making it susceptible to tipping. Because of this, the Rover will be raised to its maximum clearance only when necessary.

In order to aid in the mobility of the Rover, devices to help the astronauts avoid large obstacles, craters, crevasses, and drops were designed into the Rover. Video cameras, lights, and radar will be used by the astronauts to safely traverse the rocky Martian surface. Two video cameras

mounted on the front corners of the Rover provide visual information to the astronauts for driving near hazards that cannot be detected by the drivers looking through the windshield. The cameras will be able to rotate 360 degrees and tilt +/- 45 degrees and have zoom capability. They can also be operated manually by the astronauts. Two lights are mounted on the front of the Rover which can be rotated 30 degrees in any direction inside their sockets. The lights will illuminate the surface for any necessary night traveling and brighten the shadows behind or near objects. The Rover will use radar to aid navigation in bad weather such as dust storms. The radar is mounted in the front of the Rover.

Since Mars does not have a magnetic field good enough to use a compass for navigation, some other method of navigation will have to be devised for the Rover. One method could be to place transmitters, such as automatic direction finders (ADF), at the landing site and at the site the Rover will be traveling to. The Rover can then use the ADF signals for navigation.

Another factor that will make maneuvering on Mars difficult is the sandy terrain. Tires may not get enough traction in the sand and may have steering problems. In order to traverse the wind blown deposits expected on Mars, a very low footprint pressure of less than 0.5 lb/in^2 is needed. In order to obtain this footprint pressure, an impossibly large tire would be needed. Two alternatives to tires are loopwheels and tank treads. Loopwheels were chosen for the Rover because they are lighter than treads. They are also less susceptible to breakdown from rock and small pebbles entering the tread or loop. When loopwheels are used in a three or four wheeled arrangement, they are capable of negotiating obstacles up to a height of 75% of the loopwheel length. In order to make steering easier while traversing along slopes or over objects, the loopwheels can be operated at different levels. For example, if the Rover was traversing over rocks that were only on one side of it, only the loopwheels on that side need be raised to traverse the rocks. The Rover would remain level. This will give the Rover more stability. However, more power will be needed to drive the vehicle when the loopwheels are not at the same level.

Loopwheel sizing is a function of the ground contact area necessary for safe operations, i.e. satisfactory steering, minimum slippage, and minimum sinkage.Loose sand is the most difficult terrain to travel through, therefore this environment was used to size the loopwheels. Given the terrain (loose sand), steering angle (15°), speed (10 mph), and vehicle weight (11,000 lb), the required ground contact area can be determined. Figure 5-5 on page 653 of Reference 10* shows that the Rover must have a ground contact area

of 7000 in^2. The Rover has four separate but identical tracks. Each track is 30 inches wide and 60 inches long, giving a total ground contact area of 7200 in^2. The ground contact area was intentionally sized larger than necessary since the Reference 10* study was done with metal tracks and the Rover will use hard rubber tracks which may have more slippage since they are lighter.

The power requirements for the Rover are close approximates at best. The first assumption in calculating the power requirements is that the total weight of the Rover is 11,000 pounds on Earth or 3667 pounds on Mars. A breakdown of the Rover's weight is given in TABLE 4.4.1. To determine the frictional force experienced by the Rover, the maximum coefficient of rolling resistance was assumed to be 0.1 inches. The calculation of friction force also depends on the radius of the loopwheel, which was assumed to be 12 inches. The calculations are shown in Appendix 2. The calculations of the power required to propel the Rover on soft, level ground and up a 30 degree incline are shown in Appendix 3. For these power requirements, the effects of wind resisting the Rover's motion are assumed negligible compared to the force required to traverse a 30 degree incline. For the Rover to traverse level ground at a speed of 10 mph, the power required is 0.6 KW. The Rover requires 29.46 KW of power to climb a 30 degree incline at a speed of 4 mph. The Rover will travel over at least 280 miles and it is estimated that the Rover would be traveling for approximately 75 hours at an average speed of 5 mph. Assuming that one half of the Rover's traveling will be over rocks, the Rover will use 1162 KWH of power for transportation alone.

To supply the Rover with enough power for transportation, life support and other onboard systems, fuel cells similar to those used on the Space Shuttle will be used. These fuel cells were chosen because they have a large kilowatt per pound ratio and they are small. Each fuel cell weighs 201 pounds and is 14 inches high, 15 inches wide, and 40 inches long(Reference B*). The Rover will require three of these fuel cells to meet its power requirements. Actually, only two fuel cells are necessary for normal operating conditions at average speeds but three are necessary for peak power requirements such as traversing over rocks and up inclines. Each of the fuel cells can deliver 7 KW of continuous maximum power and 12 KW of peak power for 15 minutes. The fuel cells have a lifetime of 6700 KWH when run at an average output of 6.7 KW, Reference D*. This lifetime is more than adequate for the Rover's 20 day mission.

The power system for the Rover has a total weight of approximately 4500 pounds. The three fuel cells weigh only 603 pounds but they require hydrogen and oxygen for fuel. Each fuel cell runs off a 982 pound oxygen tank and a 308 pound hydrogen tank. These are wet weights. The total

weight for the three oxygen tanks and three hydrogen tanks is approximately 3900 pounds. The oxygen tanks are 36.8 inches in diameter and the hydrogen tanks are 45.5 inches in diameter, Reference B*. These are the outside diameters of the spherical tanks. The oxygen tanks carry excess oxygen for emergencies and life support.

The power system also requires heaters for the hydrogen and oxygen tanks, an oxygen and hydrogen flow system, a coolant loop, an electrical control unit, and a water disposal system. A water disposal system is necessary since the fuel cells produce approximately 300 gallons of water when run at 19 KW during a seven day mission. The astronauts and the radiation protection will not need all of the water produced by the fuel cells in a 20 day mission so some of it will have to be disposed of.

The size of the Rover was determined to be 12' x 7' x 8' as shown in Figures 4.4.2 and 4.4.3. Figure 4.4.3 also shows where the subsystems will be located in the Rover. The airlock and the oxygen and hydrogen tanks are on opposite corners at the back area of the Rover to obtain a weight balance in the Rover.

4.5 LABORATORY / HABITAT UNIT

4.5.1 Introduction and Groundrules

This section describes the preliminary design criteria, concepts, analysis, and systems for a Laboratory/Habitat assembly which will serve as a base for Martian surface operations. This assembly will remain on the Martian surface as part of a long range exploration plan.

1. The Lab/Hab will have some space station inheritance and will be designed to accommodate a crew of four for a minimum of one month.

2. The Lab/Hab will utilize a partially closed environmental control and life support system.

3. The Lab/Hab will be designed for easy upgrading and resupply.

4. The Lab/Hab will remain on Mars for use on return missions.

The Lab/Hab is designed to be a separate unit from the Ascent/Descent Vehicle rather than as part of the descent stage. The reasons for this decision are as follows:

a. Maximum efficiency cannot be achieved from a Lab/Hab designed around the descent stage systems and hardware.

b. Structural damage may be incurred during Ascent stage firing if the Lab/Hab is used as a platform.

c. The operational scenario may call for burial of the Lab/Hab for long term storage. This option can only be considered if the Lab/Hab is an independent structure.

4.5.2 Shape and Size

The shape and size of the Lab/Hab play an important role in its performance and cost efficiency. The criteria used in selecting the shape of the Lab/Hab were:

1. High volume/surface area ratio
2. Symmetrical configuration

3. High drag
4. Minimum necessary structural weight

The baseline shape chosen for the Lab/Hab is a cylinder of linearly varying diameter (Figure 4.5.2.1). A cylindrical shape has a higher volume/surface area ratio and a lower structural weight than a rectangular shape of equivalent volume.

The large base area increases the effective surface area of the Lab/Hab thus decreasing its ballistic number and increasing its drag. High drag is desired to aide in the deceleration of the Lab/Hab during its descent to the Martian surface. An initial base diameter of 25 ft. was chosen, giving an effective cross sectional area of 500 sq. ft. This base diameter was chosen to allow for complete stowage of the Lab/Hab into the cargo area of a heavy lift launch vehicle (30 ft. diameter assumed). The circular cross-section of the Lab/Hab maximizes the payload volume aboard the launch vehicle.

The height of the Lab/Hab is estimated to be 13 ft. This allows:

7 ft.	height of the habitation workspace
2 ft.	wall thickness, shielding, floor thickness
4 ft.	sub floor space

4.5.3 Structure and Radiation Shielding

Crew protection is essential for the safe and successful completion of the mission. The criteria for the structural and radiation shielding analysis are:

1. Ability to withstand loads during ascent from earth, descent to Mars, and impact on Mars

2. Sealed shirtsleeve housing for the crew

3. Radiation protection giving a maximum allowable dosage of 16 rems/month*

* Based on 300 rem government regulated limit for lifetime work exposure. Assuming total allowable mission exposure of (2/3)(300)=200. Assuming 12 month total mission time. 200/12=16 rems/month.

The structural approach baselined is that of a sandwich configuration. The habitation workspace is enclosed by a welded shell and a floor. This

habitation workspace shell is separated from the outer shell with a radiation insulation material of high hydrogen content. This space will also contain ECLS, power, and thermal control loop lines. The type of materials to be used for the shells will depend on g-loading, pressure loading, and thermal loading analysis.

There are two types of radiation that the Lab/Hab will have to shield against. These are:

- Electromagnetic or photonic radiation (a continuous wave type radiation whose source is the sun)

- Corpuscular radiation (particles such as electrons and protons), Cosmic radiation, the Van Allen belts, and the sun are sources of corpuscular radiation. Corpuscular radiation intensity is very high during solar flares.

The sandwich shielding of the Lab/Hab will have to stop electromagnetic and low intensity corpuscular radiation. A storm cellar of 6.5 ft. diameter will be used during periods of high corpuscular radiation. This will be located in the central sub-floor space of the Lab/Hab. In this scenario the ground crew will be informed by the orbiter crew of the oncoming event.

A shielding weight of 18,000 lbs. was obtained from the current radiation analysis. The shielding consists of aluminum and polyethylene stacks. The thicknesses required to meet the electromagnetic radiation criteria (100 ev) were 0.5" of aluminum and 3" of polyethylene. These proportions were decided on to minimize total structural weight of the Lab/Hab. The aluminum comprises the outer shielding of the Lab/Hab and is part of the Lab/Hab external structure; the polyethylene is sandwiched between the outer shielding and the inner shell enclosing the habitation workspace.

To protect against the higher corpusculent radiation released during solar flares, water will be used shield the top surface of the sub-floor storm shelter. Analysis shows that a thickness of 12" would be required to adequately protect the crew during peak periods of this radiation. The water required would be supplied from an emergency supply for this purpose and from the hygiene and wash water supply. Approximately, a total of 33 cu. ft. or 248 gallons of water will be needed to fill an enclosure of 1 ft. height above the storm cellar top surface area. Of this, approximately

100 gallons will be diverted from the existing ECLS hygiene and wash water supply.

Because of the interdependence of structural weight and shielding weight through the simulataneous use of structure as shielding, an approximation for total structural weight was made based upon the shielding weight value. The shielding weight was assumed to make up 2/3 of of the total structural weight. From this, a total structural weight of 27,000 lbs. was obtained. A total weight for the Lab/Hab is assumed to be based upon structural and shielding weight, systems weight, and expendables weight. The current total weight of the Lab/Hab is 41,000lbs. This is based upon the structural and shielding analysis and a qualitative interpolation of data obtained from references 8-10. The weight breakdown is shown below.

18,000 lbs.	Radiation Shielding
9,000 lbs.	Additional Structure (0.5 x shielding)
4,000 lbs.	Power
3,000 lbs.	Environmental Control and Life Support
2,500 lbs.	Expendables
4,500 lbs.	Science and Lab/Hab Hardware
41,000 lbs.	Total

4.5.4 Landing Analysis

In the primary proposed scenario, the Lab/Hab was to separate from the A/D vehicle at a certain terminal velocity and height. From that point the Lab/Hab would descend to the Martian surface with the aid of a parachute, deployed during separation. The candidate parachute system was the Parasail system with which the descent could be controlled. The landing of the Lab/Hab then would have been a controlled impact using a tailored composite honeycomb to absorb the energy of the impact. Since PDR-1 a FORTRAN program (LAB) was run to investigate the feasibility of this scenario with respect to acceptable impact velocities of the Lab/Hab. The program assumes a flat world and incorporates an atmosphere density profile obtained by Lincom of Houston. Input parameters of the program such as Lab/Hab weight (50,000-20,000 lbs.); parachute area (20,000-80,000 sq. ft.); separation altitude (10,000-4,000 ft.), velocity (400-100 ft/sec), and flight path angle (-70 to -50 degrees) were varied to obtain various possible terminal velocities experienced by the Lab/Hab. The results show no terminal velocities reached. . Figure (4.5.4.1) shows the velocity profile obtained by inputting the currently assumed parameters of :

Lab/Hab weight 41,000 lbs.
separation velocity 400 ft/sec
separation flight path angle -60 degrees
separation altitude 7,000 ft

and parachute area and drag coefficient of 40,000 sq ft and 1.5, respectively. The Lab/Hab reaches a minimum velocity of 260 ft/sec at an altitude of 4,200 ft. At that point the velocity begins to increase to an impact velocity of 360 ft/sec.

Figure (4.5.4.2) shows the velocity profile for the 'best' assumed case. In this, the parameters are the most extreme assumed reasonable to benefit the descent performance of the Lab/Hab. They are:

Lab/Hab weight 20,000 lbs.
separation velocity 150 ft/sec
separation flight path angle -60 degrees
separation altitude 4,000 ft.

and parachute area and drag coefficient of 80,000 sq ft and 1.5, respectively. The Lab/Hab velocity reaches a minimum of 140 ft/sec at an altitude of 3,500 ft. Then it begins to increase to an impact velocity of 290 ft/sec.

At the time of the formulation of the program, it was assumed that the deployment of the parachutes would result in the Lab/Hab reaching a terminal or near terminal velocity very near to the Martian surface. As shown, the results of the program do not agree with this assumption. Rather, the results show a deceleration to a minimum velocity sometime after the deployment of the parachute, at which there begins a period of acceleration to impact. An investigation of this discrepancy must be made. Assuming, though, that either through some fault in the computer code or with the use of additional drag devices, the minimum velocities obtained from the program can be assumed to be terminal velocities, still shows that the range of terminal speeds will not be consistent with acceptable impact speeds for energy absorbing impact devices.

From this analysis, it appears that a parachute/energy absorption device configuration alone may not be feasible as a possible landing system for use with the Lab/Hab and that a propulsion system may also have to be incorporated to obtain acceptable impact levels. This will necessarily drive major modifications to the Lab/Hab design and operational scenario and

possible to the Ascent/Descent scenario. The two recommended lines of analysis for further investigation in this case are either to provide a separate propulsion system to the Lab/Hab and keep the current A/D Vehicle-Lab/Hab descent scenario with only little modification, or, to incorporate the Lab/Hab as part of a modified descent stage and totally alter the current A/D vehicle-Lab/Hab scenario. This would obviously greatly alter the nature and design of the Lab/Hab. Of the two, the first line of analysis set forth is the more highly recommended.

4.5.5 Systems

The major systems of the Lab/Hab include:

Environmental Control and Life Support (ECLS)
Power Generation/Supply
Active Thermal Control
Airlock
Communications
Avionics

4.5.5.1 Environmental Control and Life Support Systems

Because it strikes the optimum balance between developmental costs, technological risks, and resupply penalties, a partially closed cycle ECLSS will be used in the Lab/Hab. Regenerative systems will be used for air revitalization and water-waste subsystems.

In this system, CO2 is collected and reduced to water through hydrogenation. The water is combined with humidity condensate and sterilized to form a potable water supply. Oxygen is produced through the electrolysis of water from the hygiene water supply/storage. Hygiene, wash, and urine water is reclaimed by a pre- and post-treatment phase change process. A closed water loop is provided for clothes washing in which effluent wash water is reclaimed by filtration.

The ECLS subsystems include:

Fire Detection	CO2 Removal
Thermal Control	CO2 Reduction
N2 Supply	Atmospheric control
Potable Water reclamation	O2 Generation

Potable Water storage | Waste Management
Hygiene and wash water reclamation/storage

The approximate weight of the ECLSS is 3000 lbs.
The approximate volume of the ECLSS is 300 cu. ft,
The approximate power requirement of the ECLSS is 2.5 kw

4.5.5.2 Power Systems

The criteria used in the selection of power systems were:

Primary: lightweight, reliable, high specific power
Secondary: usability for later missions

Assumption: 20 kw power will sustain Lab/Hab and science experiments.

The following power supplies were considered:

Fuel Cell/ Electrolysis unit
Multi-Hundred watt RTG
General Purpose Heat Source
Dynamic Isotope Power Systems

All data on these power sources comes from Reference 22.

FC/Eu
P: 25KW
W: 2000 lbs.
V: 400 cu. ft.
Note: Needs an external power source for the Electrolysis unit

MHW RTG
P: 150we
W: 85 lbs.
V: 2.6 cu. ft.
Note: Thermal problems and too heavy to supply necessary power

GPHS
P: 270w avg.
W: 123.5 lbs.
V: 6.6 cu. ft.
Note: Too heavy; high radiation

DIPS
P: 5KW
W: 4000 lbs.
V: 48 cu. ft.
Note: Good thermal management
10 year life

For powering the habitat systems and conducting science experiments, 4 DIPS units will satisfy the criteria .

If the surface operation scenario calls for burying the Lab/Hab, a larger power source may be needed. A SNAP-100 type thermionic direct converter could be used in this case. It would be carried down to the surface with the rover and transported to the Lab/Hab, supplying power to the rover during its trek. There, it would be emplaced at some distance from the Lab/Hab, to be used as a portable power supply. This SNAP-100 would also act as supplementary power supply for later missions.

4.5.5.3 Active Thermal Control

To provide proper temperature environments for the proper functioning of the crew and equipment aboard the Lab/Hab, thermal control systems must be used These include both active and passive systems.

The ECLS has an active thermal control subsystem to modulate the internal atmosphere of the crew. In addition to this, active thermal control must be provided in various subsections of the Lab/Hab to insure proper functioning of the systems. Heat dispersion must be facilitated in various areas of the Lab/Hab subject to thermal buildups from operating subsystems. Most prominent of these is the power production subsystem which will produce peak temperatures up to 360 degrees Centigrade. With the use of a thermal management system, such areas can be used as heat sources or sinks, as may be required.

In addition to active thermal control, passive thermal control must be provided. The use of thermal insulation materials and heat rejection coatings will be critical to thermal control during periods of peak positive and negative heat gradients through the inner and outer shells of the Lab/Hab.

4.5.5.4 Airlocks

Two airlocks will be necessary to provide the transfer of crew and equipment into and out of the Lab/Hab. The first is a main airlock, the second is a service airlock.

Four people will be continuously manning the Lab/Hab and it is assumed that at each shift they will divide into (two) crews of two; one crew inside the Lab/Hab and one crew performing external surface activities. To design for this, the main airlock will allow a crew of two to simultaneously enter and exit the Lab/Hab. This was chosen as the most efficient arrangement with respect to total power requirements, total depress-repress time, and total air pressure losses. To minimize its volume, the main airlock

will not provide area to stow gear and life-suits. It will however provide apparatus to externally clean the life-suits and partially sterilize them. The main airlock will also be used to accommodate high bulk equipment, tools and science apparatus once the external surface activity crew is outside. The main airlock will perform depress to outer hatch opening in no more than five minutes. An approximate volume for the main airlock is 180 cu ft. Based on ref. (8), the main airlock's weight will be approximately 1,000 lbs. and will use 1.5kw during operation.

Apart from the main airlock, a smaller service airlock will be used to allow for the transfer of smaller equipment and tools to the crew outside. This airlock will have an approximate volume of 12 cu ft., weigh 70 lbs. and use 100w of power during operation.

4.5.5.5 Communications and Avionics

No detailed baselines for the communications or avionics subsystems have as yet been made. The communications subsystem will generally follow the uplink/downlink diagram of figure (4.4.1) and provide for the high-speed transmission, receiving and processing of voice, telemetry, commands, data, and video.

4.6 UNMANNED SURFACE EXPLORATION

For unmanned exploration of the surface of Mars, Phobos, and Deimos, two different types of probes will be used. Several Penetrators will be deployed to Mars, and a non-recoverable Soft Lander will be sent to each of the moons. The only difference between the two Soft Landers is the amount of fuel required to reach their destinations.

The goal of these probes is to gather scientific data in five areas. These areas are geochemistry, water detection, heat flow, stratigraphy, and imagery. It is expected that this information will aid scientists in their explanation of the formation, evolution, and present state of Mars and its moons. Because Mars has many different types of surface configurations, it is desired that most of the configurations be examined in order to optimize the scientific return. Therefore, twenty Penetrators have been recommended for this mission.

4.6.1 Penetrators

The concept of using Penetrators to gather scientific information is not a new one. The U. S. Army used Penetrators in the Vietnam War, thus establishing the Penetrator as a feasible and reliable system.

The major advantages of using several Penetrators on Mars are:

1) To allow the establishment of a network of information gathering stations on Mars. This will allow the following phenomena to be measured: a) Mars quakes, b) Seasonal variations in the atmospheric processes, c) Locations of the magnetic fields.

2) To enable experiments to be conducted beneath the Martian surface.

3) To allow measurements of geochemical, heat flow, and geophysical observations on the Martian surface that could not be reached by larger, more sophisticated soft landers.

4) To provide information for selecting sites for future missions to Mars.

The Penetrator system consists of four major components. These components are the launch tube, deployment motor, aero decelerator, and the surface penetrator containing the scientific payload. The maximum system weight is 166 lbs. This weight allows a contingency weight of 8 lbs.

Also, the deployment motor, aero decelerator, and the Penetrator with its payload is stored in and launched from the launch tube. The launch tube will be capable of being hermetically sealed to insure that the systems will remain sterilized in order to minimize contamination of the Martian surface. The overall size of this system is dictated by the size of the launch tube, 12 inches in diameter by 92 inches in length.

The scientific payload each Penetrator will carry consists of the equipment required to conduct the following experiments:

1. Seismic
2. Magnetic
3. Geochemical
4. Water detection
5. Stratigraphy
6. Imaging
7. Meteorologic
8. Heat flow

Twenty Penetrators will be deployed over a wide region of the Martian surface. This will allow two Penetrators to be sent to each site except those in the plains: Acidalia Planitia, South Polar Region, Syria Rise, and Arsia Mons. The remaining eight sites have either heavily cratered or chaotic terrain that could damage the impacting Penetrator, therefore an extra Penetrator has been allotted to these regions. The possible impact sites do not include the areas where Vikings I and II landed nor does it include the scheduled sites for the manned landings. The selected site are as follows:

	Site Name	Latitude	Longitude
1.	Acidalia Planitia	46° N	5° W
2.	Amphitrities Patera	59° N	297° W
3.	South Polar Region	81° N	190° W
4.	Syria Rise	15° S	105° W
5.	Cratered Highlands	30° N	10° W
6.	Hellas Rim	29° S	303° W
7.	S. Polar Cratered Region	68° S	160° W
8.	Canyon lands	10° S	73° W
9.	Arsia Mons Plains	8° S	130° W
10.	Cratered Uplands	25° N	150° W
11.	Argyre Planitia	50° S	40° W
12.	Isidis Planitia	15° N	270° W

Figure 4.2.1 shows the locations of these sites on the surface of Mars.

There are three different ways to arrive at the suggested impact sites. They are 1) launch the Penetrators from the Martian surface, 2) launch the Penetrators from the orbiter while in a circular orbit, 3) launch the Penetrators from the orbiter several days before orbit insertion.

Launching the Penetrators from the surface should be avoided, as this will require excessive fuel demands proportional to the total distance the Penetrator must travel. Launching the Penetrators from the orbiter, while in a circular orbit, would also require excessive amounts of fuel, because this maneuver would necessitate out of plane changes of 10 degrees or more for 11 of the 12 suggested impact sites. The third method, launch several days before orbit insertion, has the advantage of lower fuel requirements. For this method, the required change in velocity to place a Penetrator on a trajectory to arrive at one of the scheduled impact sites is inversely proportional to the distance from Mars to the Penetrator. Thus, the required velocity increase will reduce as the distance between Mars and the Penetrator is increased. Therefore, it is recommended that the Penetrators be launched towards their designated impact site while the orbiter is on its hyberbolic approach trajectory. Because this type of maneuver requires only a small velocity change, the Penetrators have been designed to develop a maximum delta-V of only 130 ft/sec. (This velocity change is provided by five pounds of solid fuel with an I_{sp} of 260 seconds.) This maximum delta-V is sufficient to place a Penetrator on a trajectory to any of the suggested impact sites.

When the Penetrators are launched from the orbiter, they will have an approach velocity of 14,000 ft/sec. Because of the sensitivity of the scientific payload, each Penetrator must impact the surface of Mars under 525 ft/sec. Impacting under this velocity will insure that the integrity of the instruments is maintained. Two aero decelerators will be used to slow each Penetrator.

The first decelerator ia a hypersonic decelerator. This is a large cone shaped structure with a half-cone angle of 60 degrees when it unfolded from the Penetrator's body. This decelerator will be deployed when the Penetrator enters the Martian atmosphere and will be jettisoned approximately 5 miles above the surface. At this point, the hypersonic decelerator will have reduced the initial velocity of 14,000 ft/sec to 940 ft/sec. This change in velocity was determined by using the equations of motion and assuming a coefficient of drag of 0.92.

After the hypersonic decelerator is jettisoned the final, or terminal, decelerator is deployed. This decelerator consist of two square drag plates that will unfold from the Penetrator's body near the rear section. These drag plates will be positioned so as to minimize any unwanted pitching moments. Assuming these plates have a cross-sectional area of 0.45 square feet and a coefficient of drag of 0.62, they will slow the Penetrator to 484 ft/sec upon impact with the surface. After the Penetrators impacts the surface, they are ready to begin gathering the scientific data to be transmitted back to the orbiter. Figure 4.6.1.1 diagrams the Penetrator system.

4.6.2 Soft Landers

Two non-recoverable unmanned Soft Landers will be used to explore the moons of Mars. The only difference between the two Landers will be the amount of fuel required to get them to their destination and the size of their fuel tanks.

Each Lander will carry the following instruments as its primary payload:

1. Seismometer
2. Heat flow thermocouples
3. Proton backscatter and X-ray fluorescence
4. Accelerometer
5. Magnetometer
6. Meteorology equipment
7. Imager (camera).

This primary payload weighs 17 lbs. To support the scientific instruments the following subsystems will be needed:

1. Nickel- Cadmium batteries for power,
2. A transmitter to relay all findings back to the orbiter,
3. A receiver to receive commands from the orbiter,
4. A structure to house the payload and the subsystems,
5. Three control jets to maneuver the Lander during the landing phase,
6. A liquid monopropellant precision controlled rocket engine.

The weight of the Lander, including subsystems and scientific payload, is 227 lbs. This weight does not include the required amount of fuel to propel each Lander, or the weight of the fuel tanks needed to store this fuel.

In order to determine the fuel weight, the required velocity change is needed. This velocity change was determined by using the Hohmann

transfer method and assuming the moon's orbits were in the same plane as the orbiter. Also, it was assumed that the Landers were launched from a 270 nautical mile parking orbit and were using hydrazine fuel. From these assumptions, the required ΔV for Deimos was computed to be 5430 ft/sec and for Phobos the required ΔV was determined to be 3704 ft/sec.

With this information, along with the density of hydrazine (4.379 lbs/cu. in.) and aluminum 2014-T6 (0.101 lbs/cu. in.), an iteration was done to determine the weights of the required fuel and fuel tanks for each Lander. For the Deimos Lander 877 lbs of fuel is needed along with a 734 lb fuel tank. For the Phobos Lander, 370 lbs of fuel and a 436 lb fuel tank are needed. Therefore, the total weight of the Deimos Lander is 1840 lbs and the Phobos Lander weighs 1033 lbs. Figures 4.6.2.1 and 4.6.2.2 illustrate the overall sizes of each Lander.

When the Landers are launched and have completed their orbit transfers, they will be positioned between 165-330 feet behind each moon in a tail chase. The velocities of the Landers will be the same as the velocities of the moons they are following. The control jets will be used to accelerate and control each Lander for landing on the moons. While the Landers are on the moons, they will be able to rotate 360° on their landing gear to allow the imager to take photographs in any direction. The imager direction will be controlled by using the control jets.

Although the gravity on Phobos and Deimos is extremely small, the Soft Landers are not equipped with any type of system to fasten them to the surface. The nature of the experiments to be conducted on the moons do not require that the Landers be fastened. If it becomes necessary to fasten the Landers to the moons, this can easily be incorporated into the proposed system.

5.0 Management and Cost

5.1 Organizational Structure

The organizational demands of the Mars Manned Landing project differ from a typical engineering project in that the members of the project are students, and the project is set in a academic environment. The following four points are the special organizational demands imposed on the project.

1. The average level of experience for individuals on the project is low.

2. The project schedule must fit into the time frame of one semester.

3. The week to week manpower available to the project is highly variable.

4. Individuals on the project are responsible to the project for a grade only. Individuals on the project are not paid employees.

The project has been divided up into four groups: the Mission Planning group, the Martian Operations group, the Management group, and the Quality Control group. The overall organization of the project was structured to intuitively match the structure of the problem at hand, to insure quality work, and to provide responsive communication to our contract monitor. Figure 5.1.1 illustrates the project's organization and the personnel assignments.

The technical portion of the project has been divided up into two groups: the Mission Planning group, and the Martian Operations group. The two groups were created in order to limit groups to approximately seven persons each, and to simplify the management requirements of the technical managers. The two groups were defined so as to reduce the amount of coordination required between the groups and to allow each group to specialize in their own class of technical problem.

Due to the inexperience of the members of the project, quality control is an important issue. Therefore a separate quality control group has been created. This group observed the activity of the other groups, from a critical view point, in order to challenge the operation of the groups and to introduce alternate approaches. This group is manned by the teaching assistant, and the project manager. These individuals are considered the most experienced persons in the project and are best able to provide technical guidance.

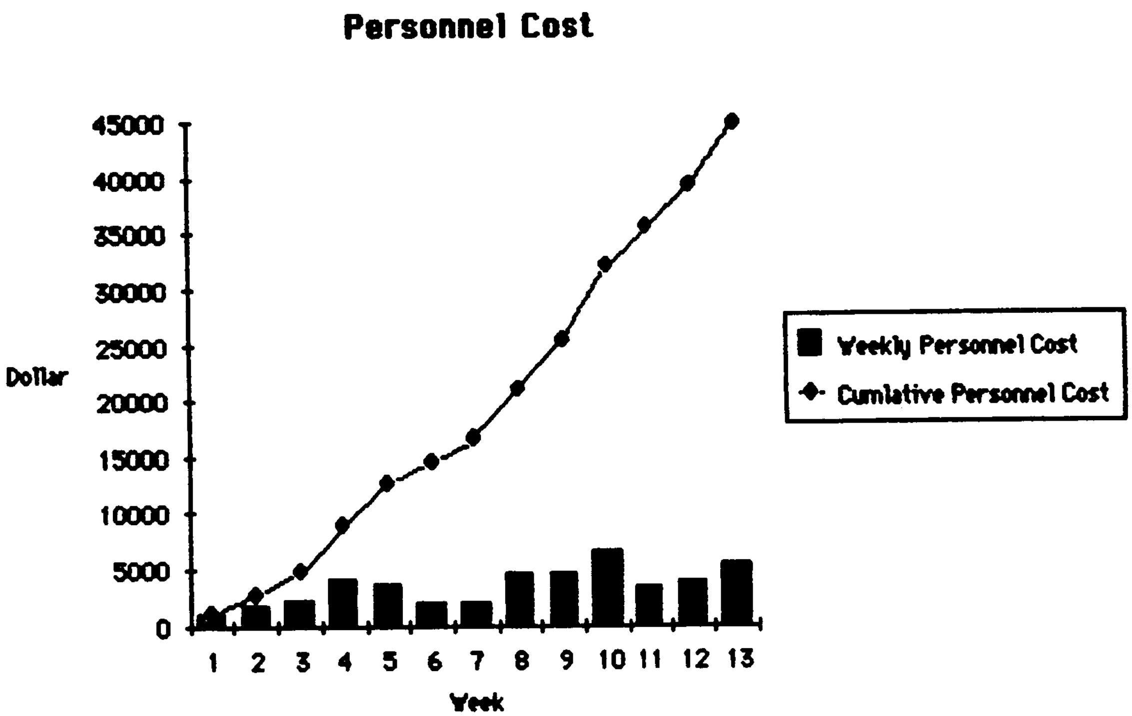
A management group has been created to take charge of the week to week tracking of the project's status. This task supported the project manager in reporting the project's status to the contract monitor, and generated the management data required for the various data items defined in the RFP (Request For Proposal).

From time to time during the project, action items or milestones were required for the creation of a short term action group. The project manager was responsible for creating these groups when the situations occurred.

The management organization was structured to provide reasonable span of control, unity of command and effective communication between the managers. The managers of the four groups report directly to the project manager, who in turn reports directly to the contract monitor. As a rule, there are no management positions below the four specified group managers. The project is perceived too small to warrant any additional management complexity.

5.2 Personnel Costs

The personnel cost was based on the actual hours accumulated from the beginning of the project to the present. The GS federal pay scale was used as a basis with the following breakdown within the organization: Engineers were GS-13 with a rate of $37599 per annum, First line supervisory management were GS-14 with a rate of $44430 per annum, and the program manager was a GS-15 with a rate of $52262 per annum.



From the hours recorded, it was seen that the hours increased in the week before delivery of the Conceptual Design Review, PDR1 and PDR2.

5.3 Material and Hardware Cost Estimate

The Government Furnished Equipment (GFE) consists of computer hardware, software, mainframe computer time and supplies as furnished by the University of Texas at Austin.

One Time Costs

1 Macintosh,Printer,External Disk Drive, Modem,Software	$2697.00

The costs incurred buying the Macintosh computer, peripheral devices, and software are actual costs.

Total Costs for the Project

Personnel (Manhours)	$45,000.00
Material Costs	$2697.00
Grand Total	$47,697.00

6.0 Figuires/Tables

4.6.1.1 Penetrator Systems and Launch Tube Assemblies

4.6.2.1 Deimos Lander

4.6.2.2 Phobos Lander

Table 2.1.1.1 COMPONENT SPECIFICATIONS

A.) Propulsion and Power	
Power	40 MW
Fuel	Uranium mononitride
Radiator:	
Area	3880 ft^2
Volume	6000 lb
Propellant (LEO):	
Weight	1,152,000 lb
Volume	1,356.9 ft^3
Engine:	
Power	11.3 MW
I_{sp}	5800 sec
Thrust	90 lb
Attitude Control:	
Propellant weight	19,000 lb
System hardware	16,000 lb
Component weight	1,193,000 lb
B.) TDRS Satellite System	
4 TDRS satellites w/IUS	120,000 lb
Canisters	12,000 lb
Component weight	132,000 lb
C.) Martian Operations	
Component weight	300,000 lb
D.) Structure	
Core, shadow shield, rings	300,000 lb
Enviromental area, LSS	100,000 lb
Component weight	400,000 lb
TOTAL VEHICLE WEIGHT	**2,025,000 lb**

TABLE 3.1

Parameter Values Used in Parking Orbit Analysis

CAPTURE	$M_{dry\ barge}$ = 250,000 lb.
	I_{spc} = 450 sec.
	$V_{hyper} = V_{\infty}$ = 8700 ft./sec.
ASCENT	$M_{dry\ AV}$ = 5,000 lb.
	I_{spa} = 260 sec.
DESCENT	$M_{dry\ DV}$ = 24,000 lb.
	I_{spd} = 260 sec.
	M_{rover} = 10,000 lb.
	$M_{lab/hab}$ = 35,000 lb.
GENERAL	G_0 = 32.2 ft./sec.^2
	μ = 1.515E15 ft.^3/sec.^2
	r_{mars} = 10,876,219 ft.

1. DOCKING INTERFACE/AIRLOCK
2. REACTION CONTROL THRUSTERS
3. ASCENT/DESCENT VEHICLES
4. ENVIRONMENTAL CORE
5. AIRLOCKS
6. ANTENNA
7. ENVIRONMENTAL VESTIBULE
8. PENETRATOR STORAGE, OBSERVATION EQUIPMENT
9. COMMUNICATIONS, FLIGHT CONTROL SYSTEMS
10. REACTION CONTROL PROPELLANT STORAGE
11. MARTIAN OPERATIONS PROPELLANT STOODRAGE
12. TRACKING AND DATA RELAY SATELLITE SYSTEM STORAGE
13. LIQUID MERCURY STORAGE
14. COOLANT LOOPS
15. REACTOR, SHADOWING CONE
16. LOW THRUST ENGINE NOZZLE
17. MOON LANDERS

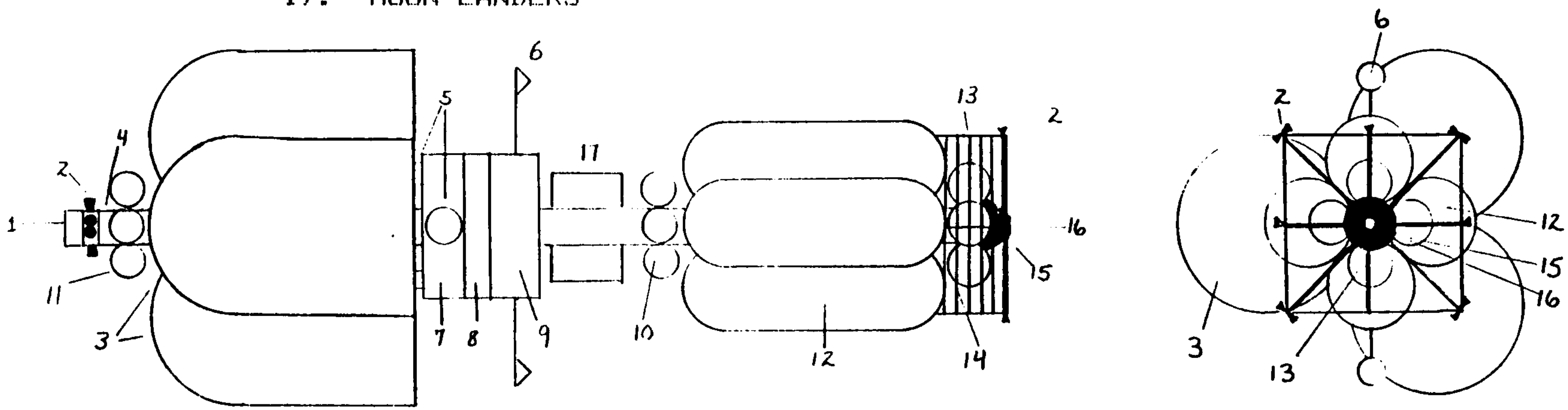
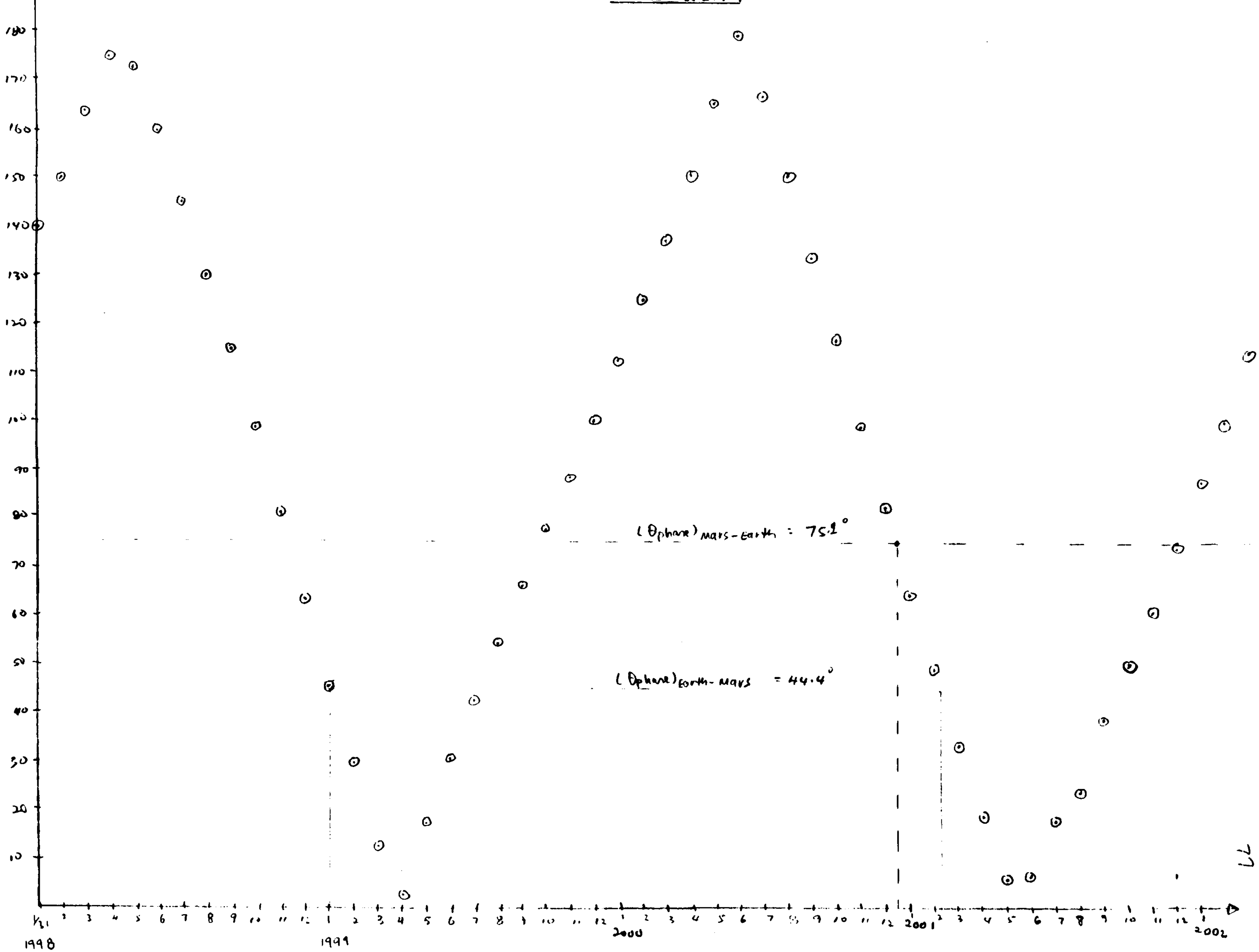


Figure 2.1.1 Vehicle Lay out

FIGURE 2.3.1.1

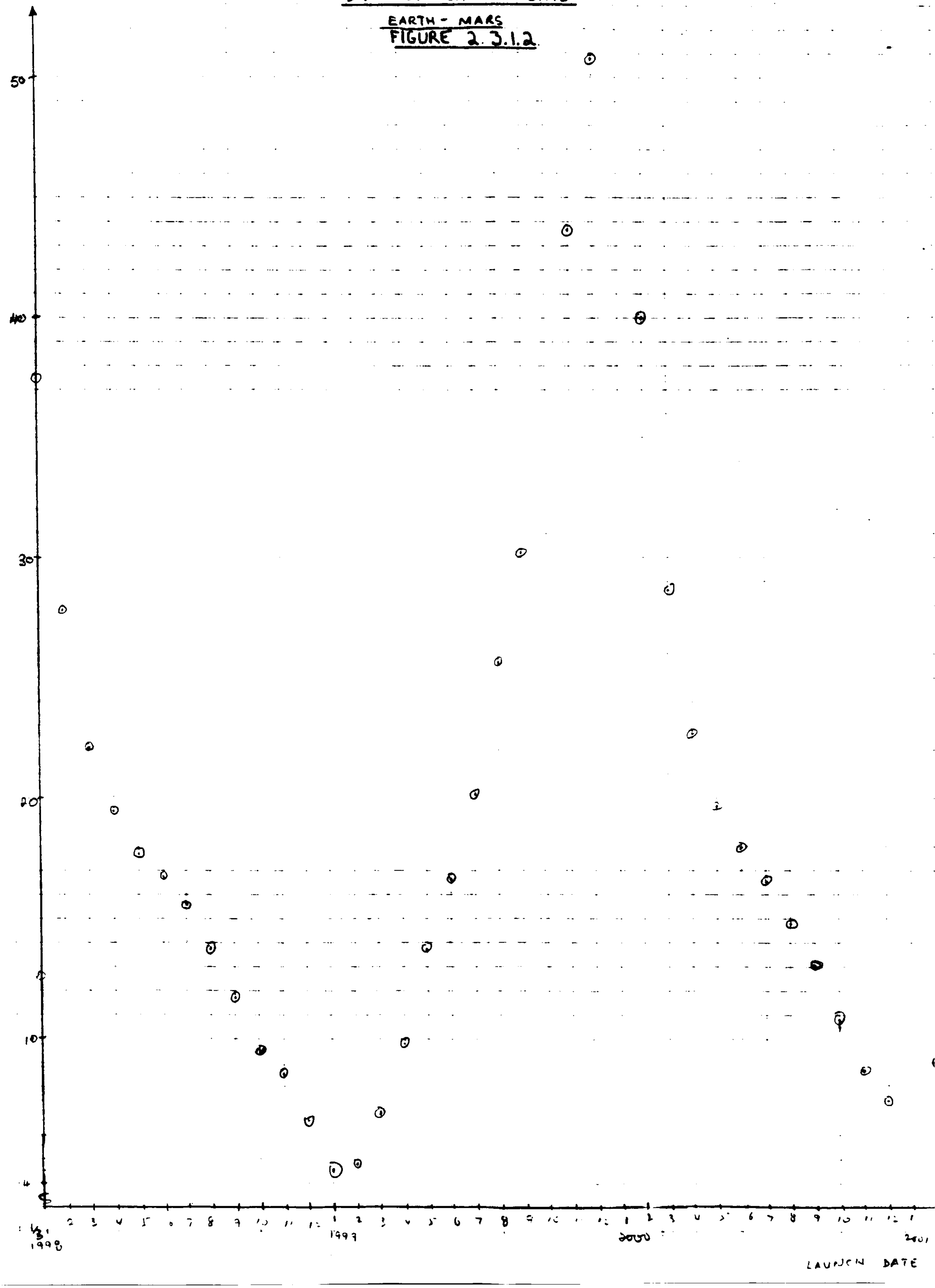


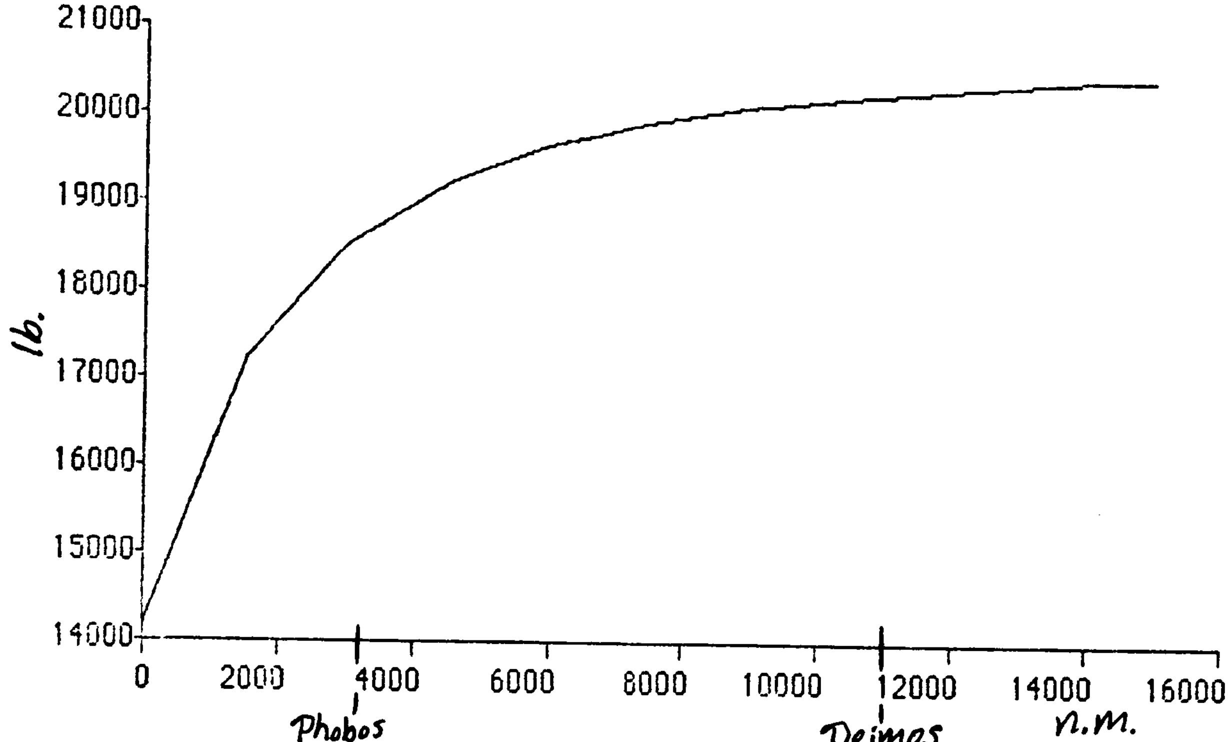
180
170
160
150
140
130
120
110
100
90
80
70
60
50
40
30
20
10
$(\theta_{phase})_{Mars-Earth}$ = 75.2°
$(\theta_{phase})_{Earth-Mars}$ = 44.4°
1998
1999
2000
2001
2002

ΔV VS. LAUNCH DATE

EARTH - MARS

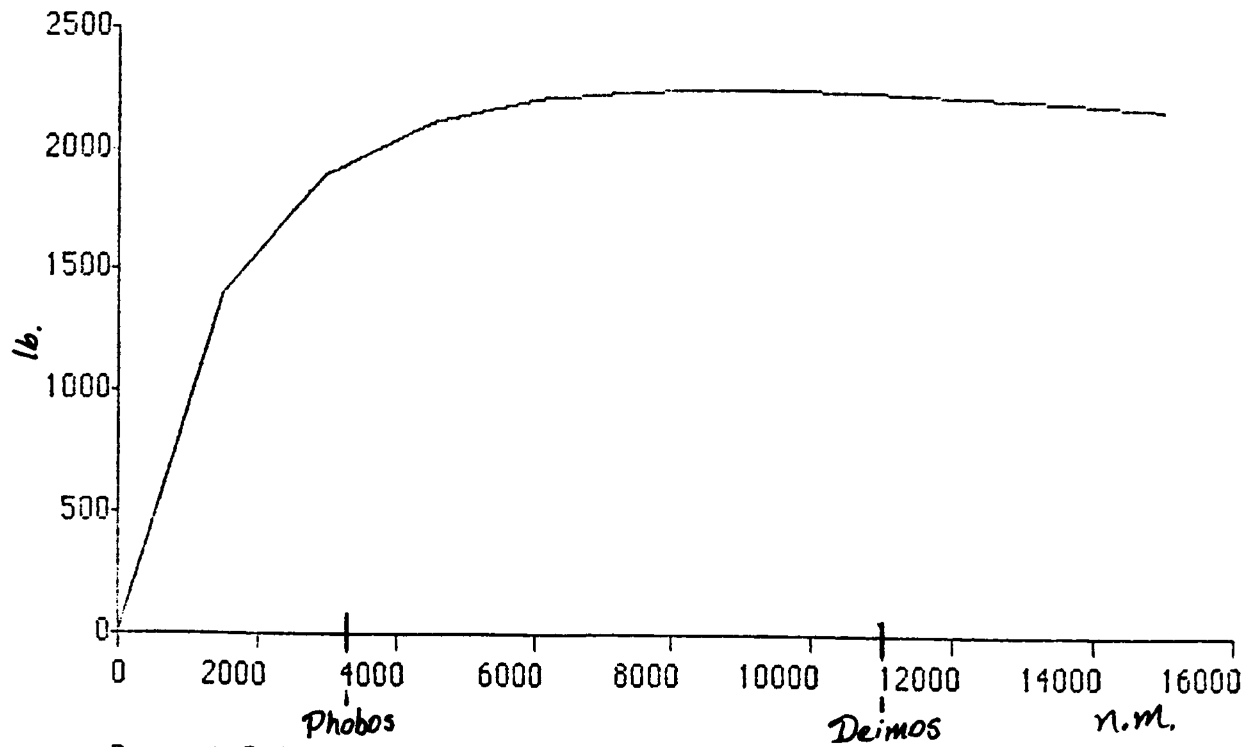
FIGURE 2.3.1.2





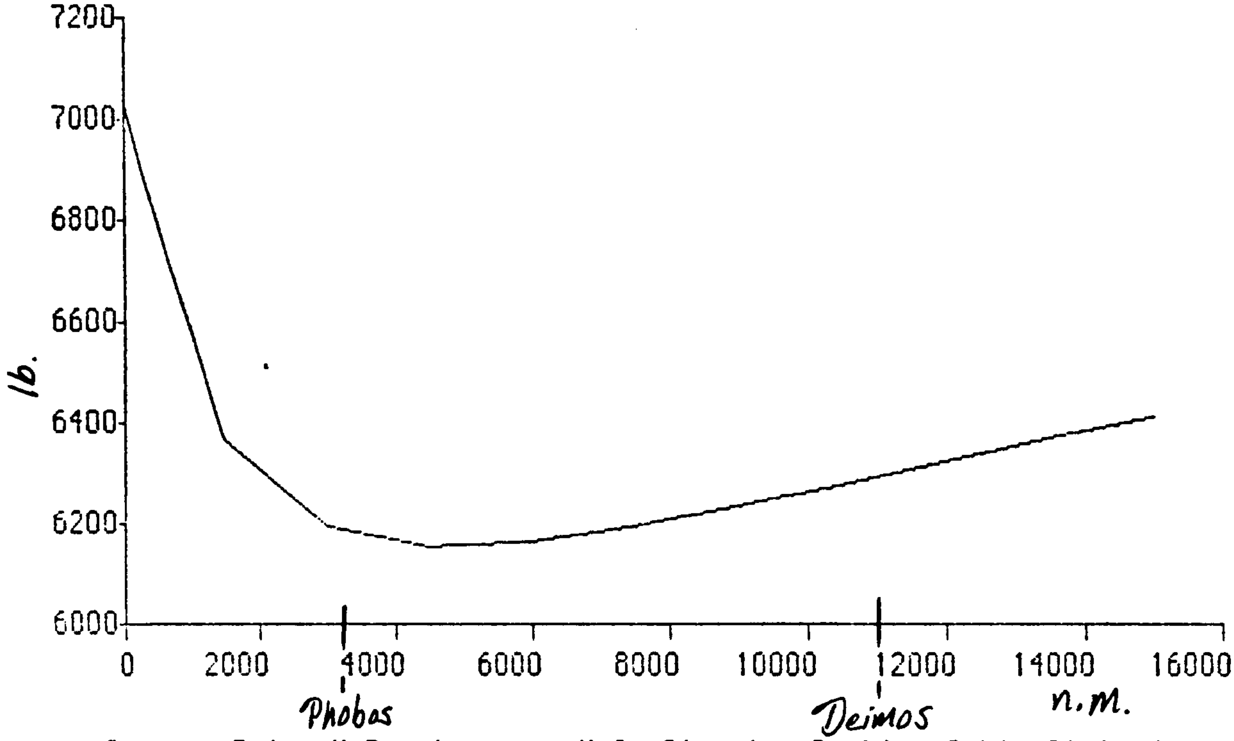
Ascent Delta V Requirements V.S. Circular Parking Orbit Altitude

FIGURE 3.1



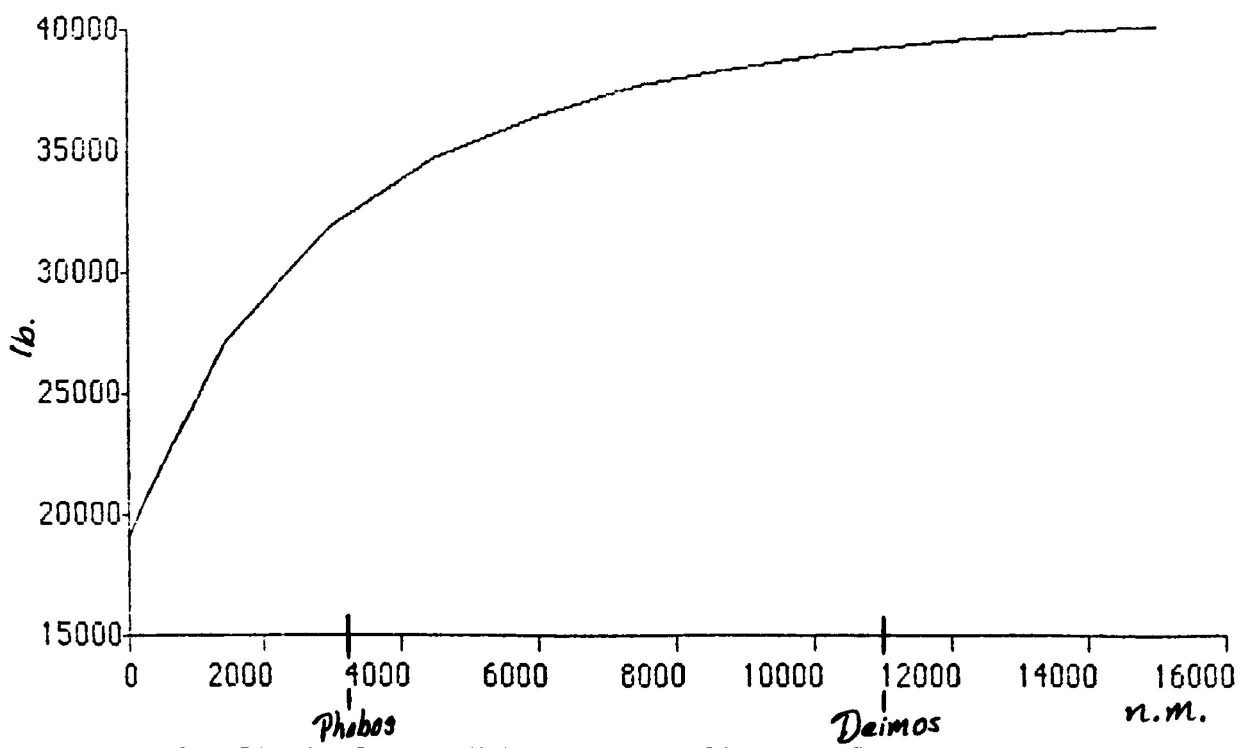
Descent Delta V Requirements V.S. Circular Parking Orbit Altitude

FIGURE 3.2



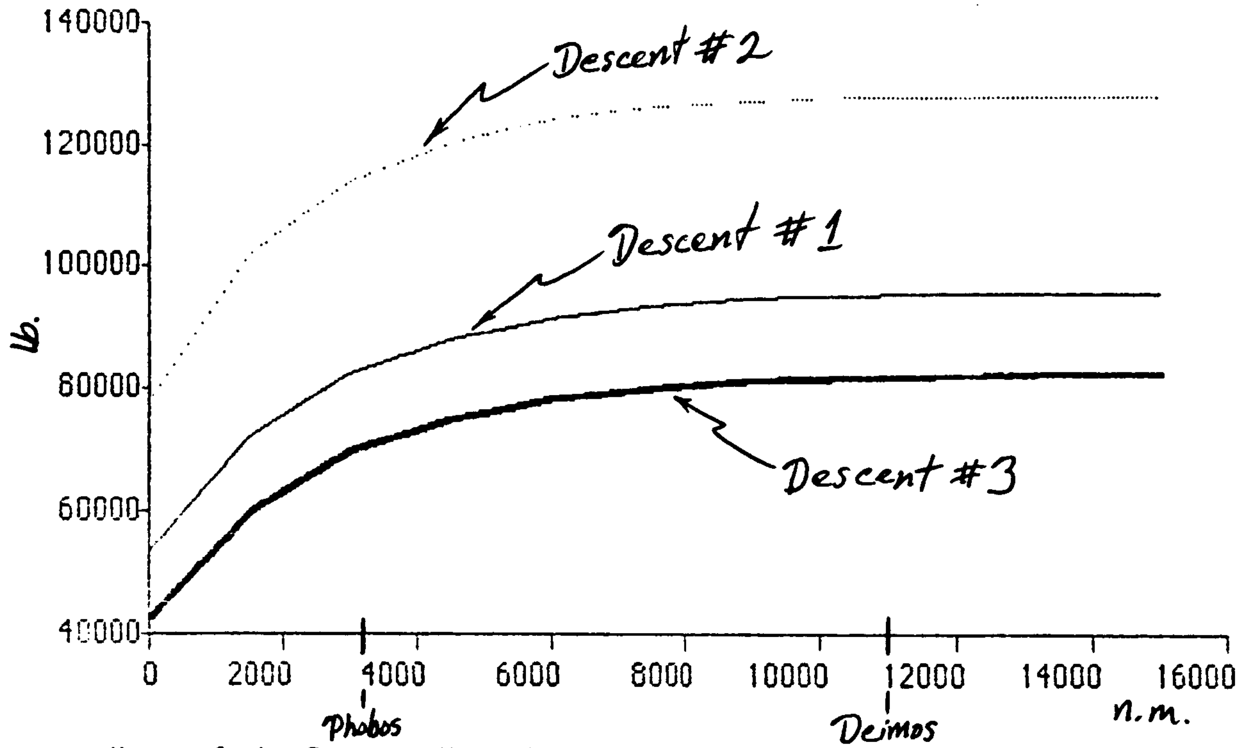
Capture Delta V Requirements V.S. Circular Parking Orbit Altitude

FIGURE 3.3



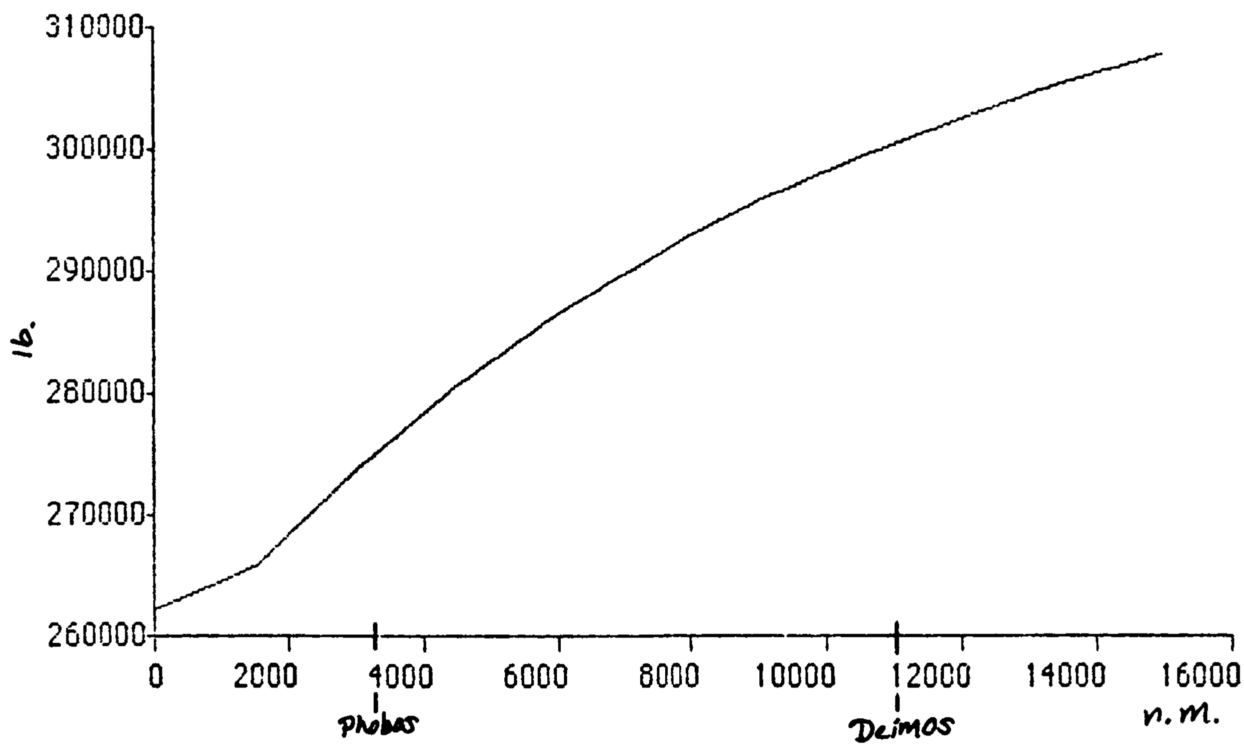
Mass of a Single Ascent Vehicles V.S. Circular Parking Orbit Altitude

FIGURE 3.4



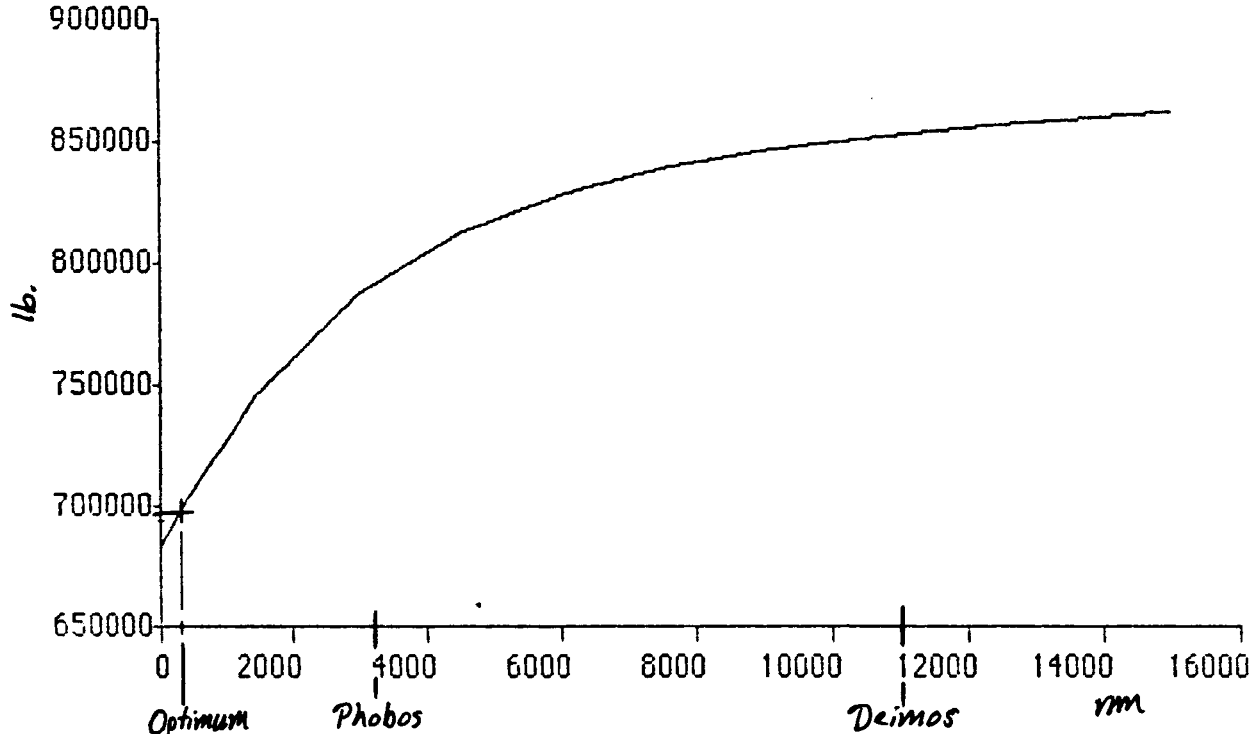
Mass of the Descent Vehicles V.S. Circular Parking Orbit Altitude

FIGURE 3.5



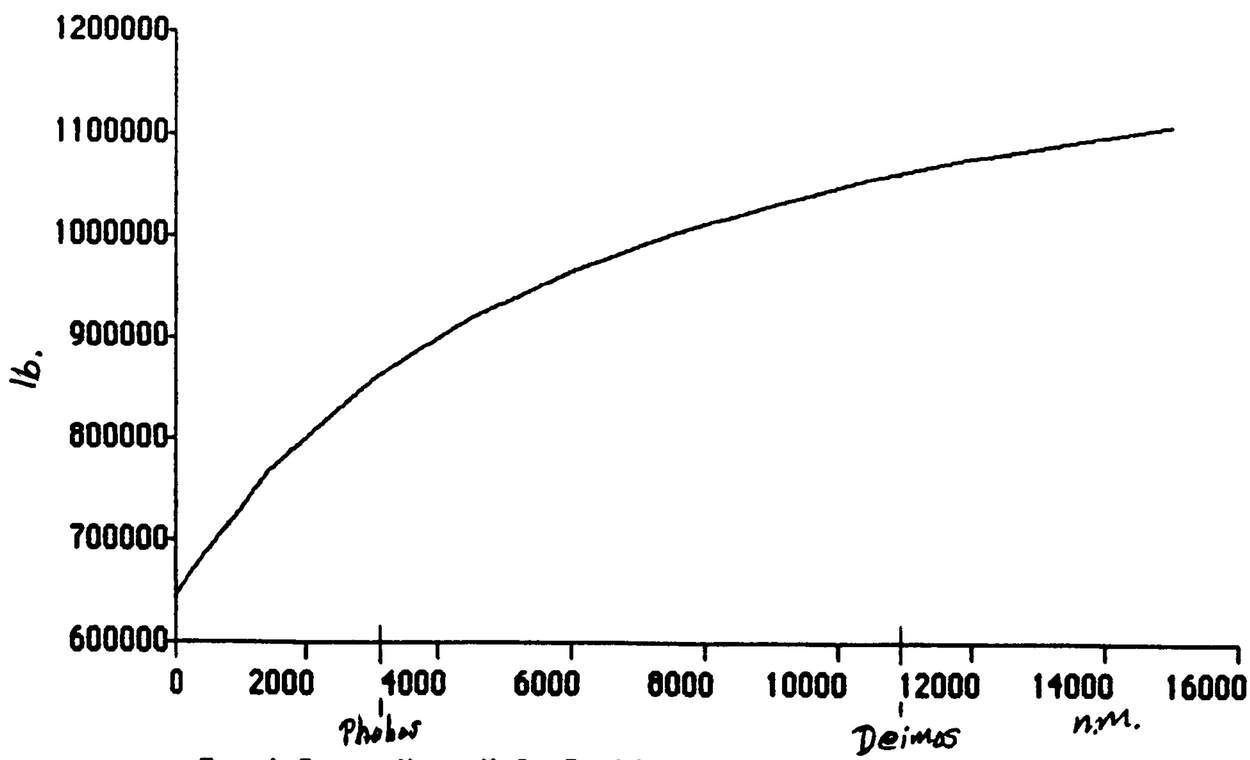
Mass of the Capture Fuel V.S. Circular Parking Orbit Altitude

FIGURE 3.6



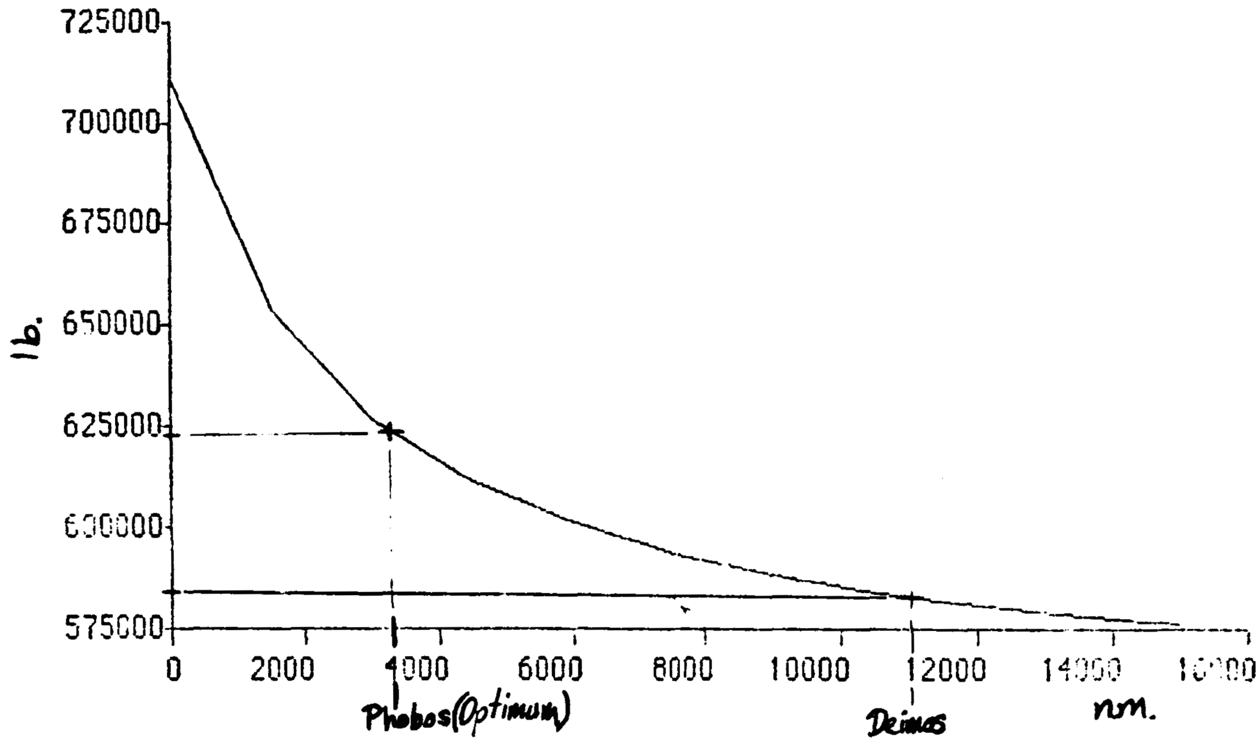
Total Barge Mass U.S. Circular Parking Orbit Altitude

FIGURE 3.7



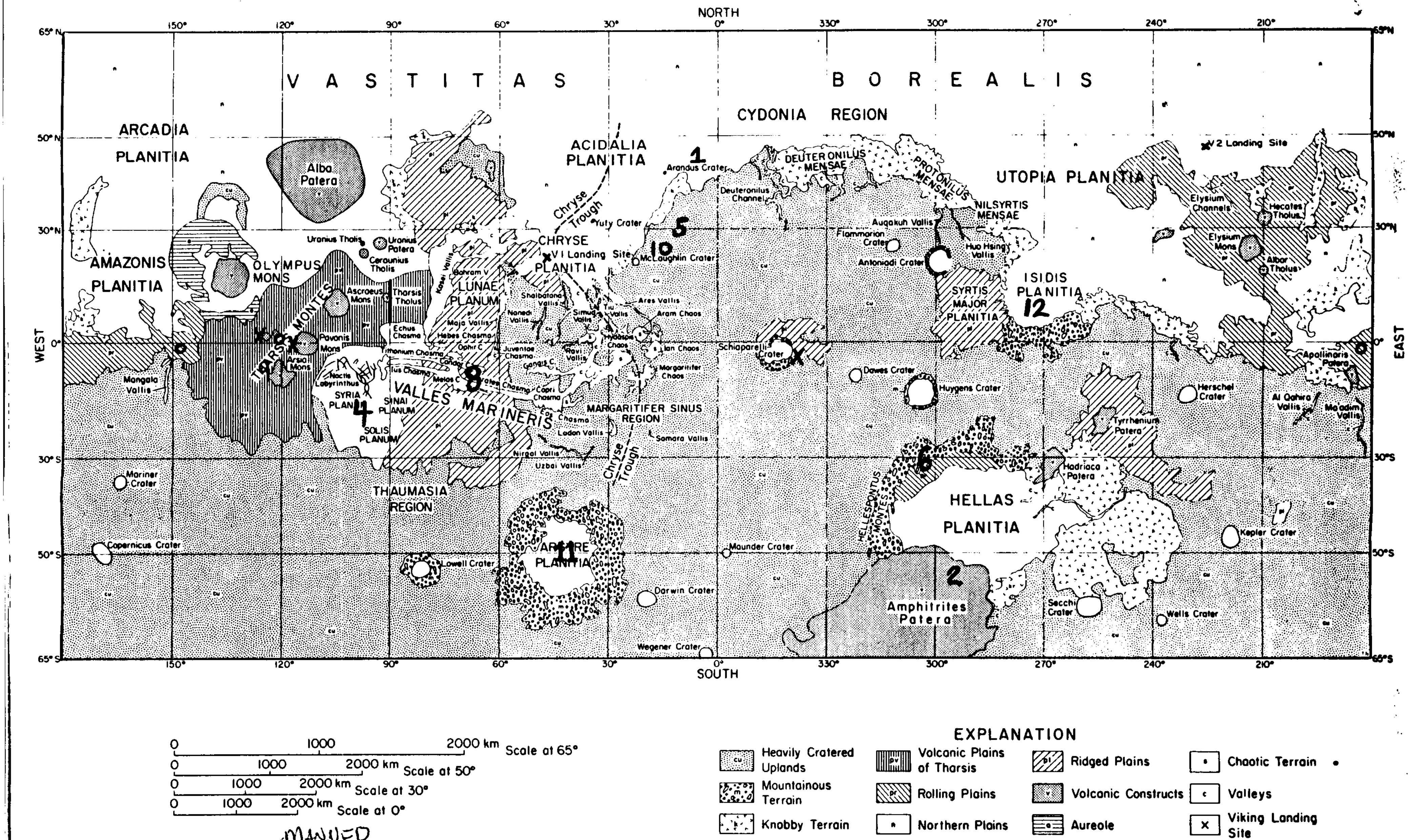
Total Barge Mass U.S. Parking Orbit Perigee Altitude

FIGURE 3.8



Total Barge Mass V.S. Parking Orbit Apogee Altitude

FIGURE 39



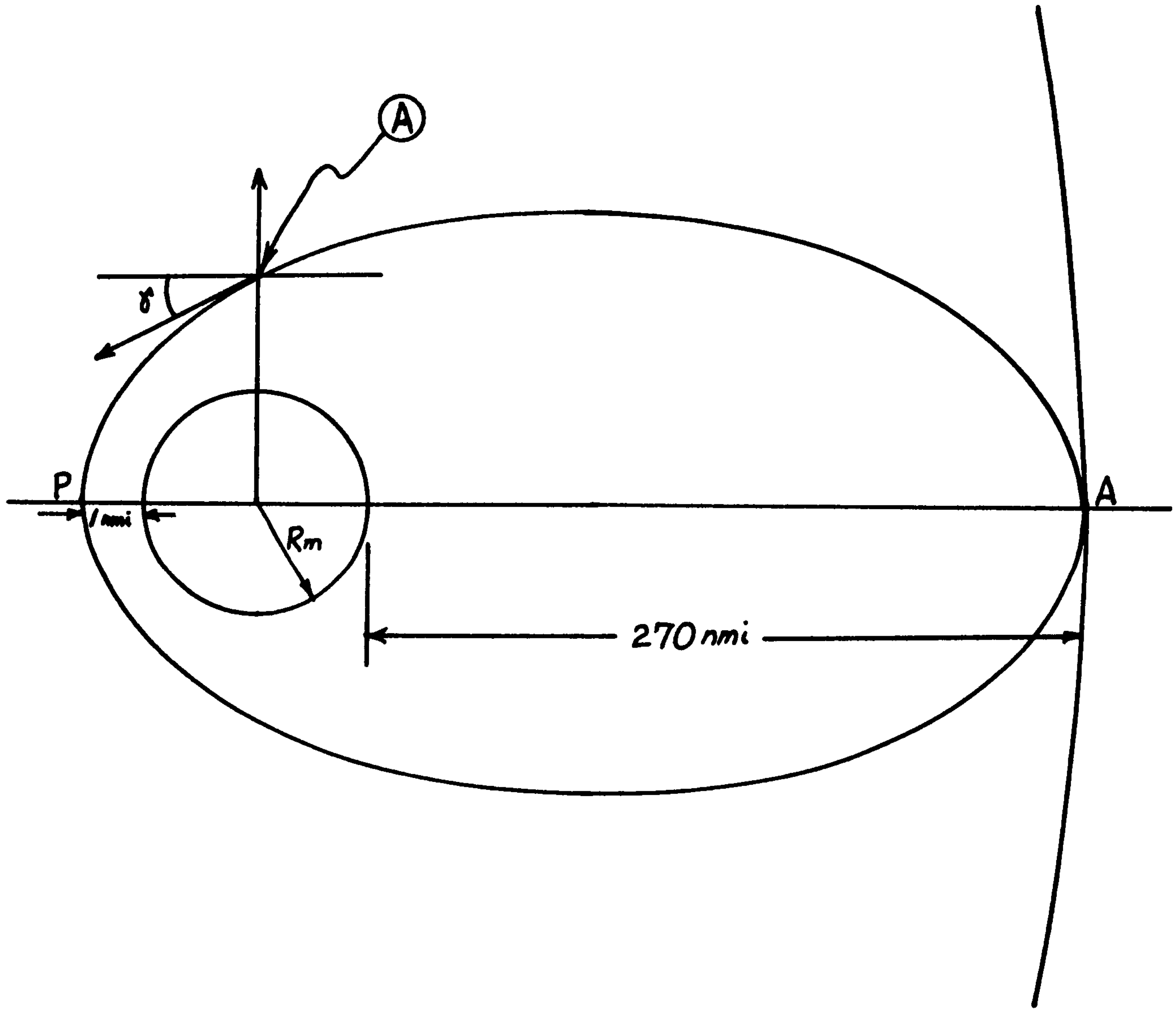
X- PRIMARY MANNED LANDING SITES

O- OPTIONAL MANNED LANDING SITES

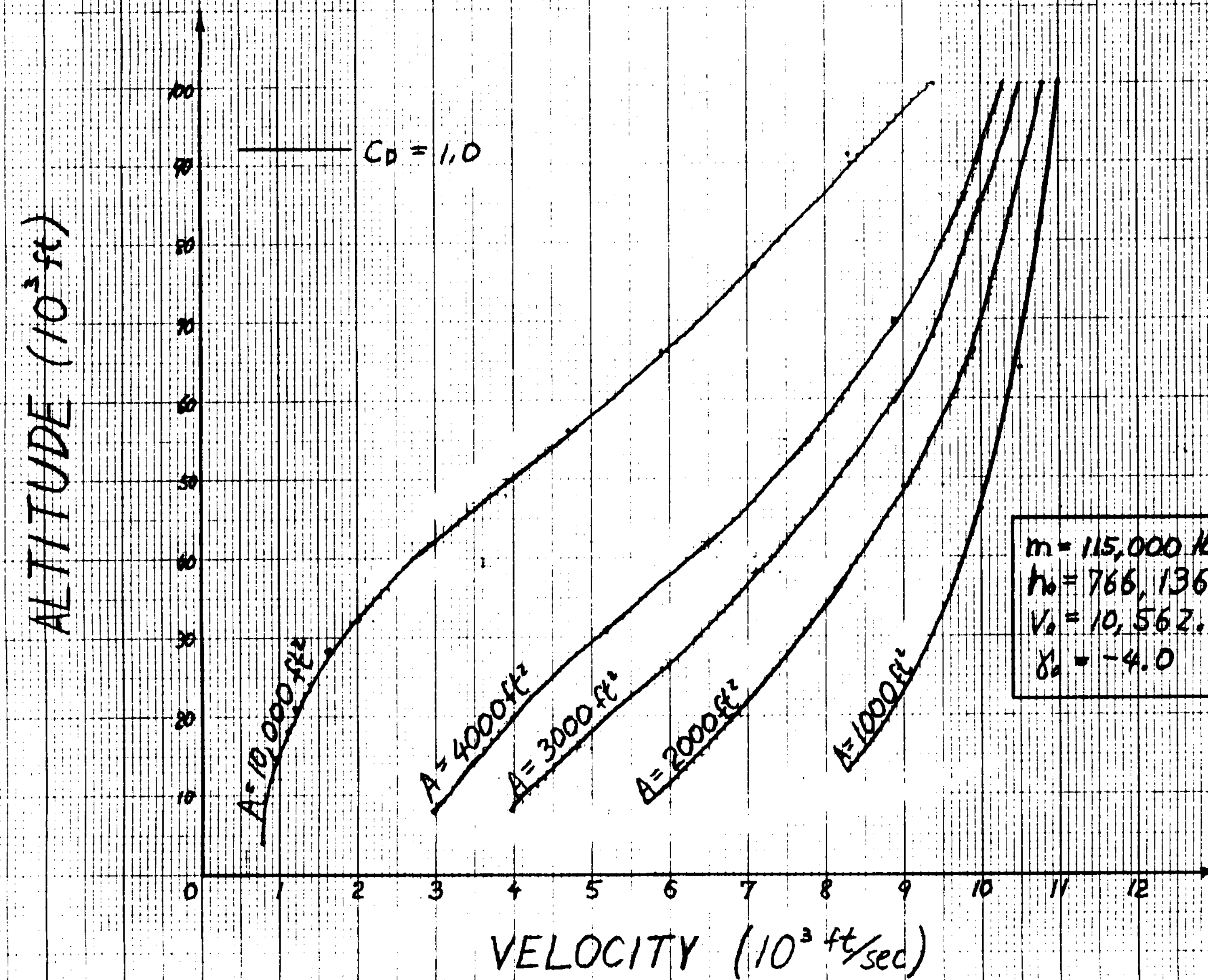
numbered locations represent

LANDING SITE LOCATIONS

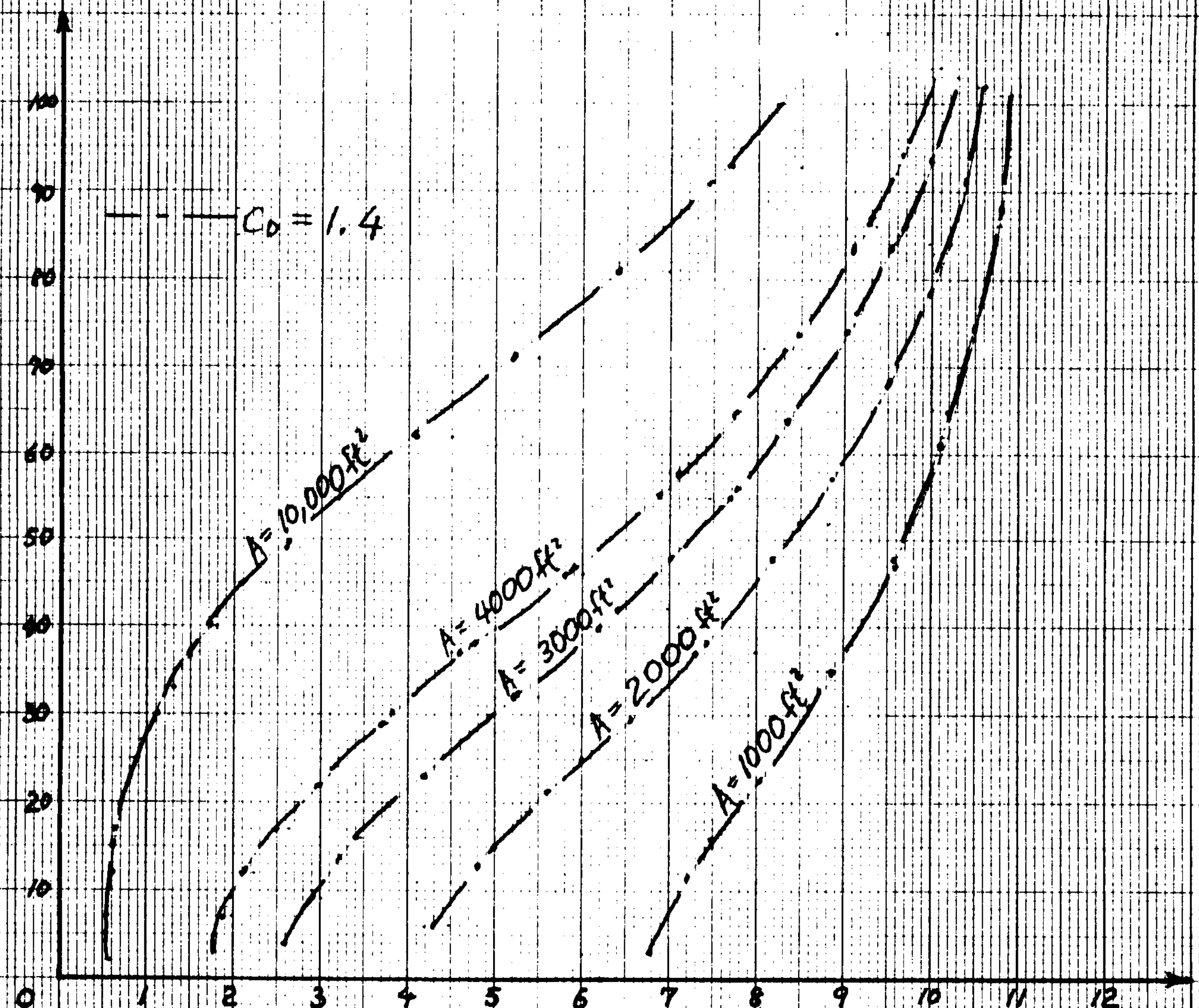
FIGURE 4.2.1



INITIAL PARAMETERS FOR DESCENT VEHICLE ANALYSIS
FIGURE 4.3.3.1.1

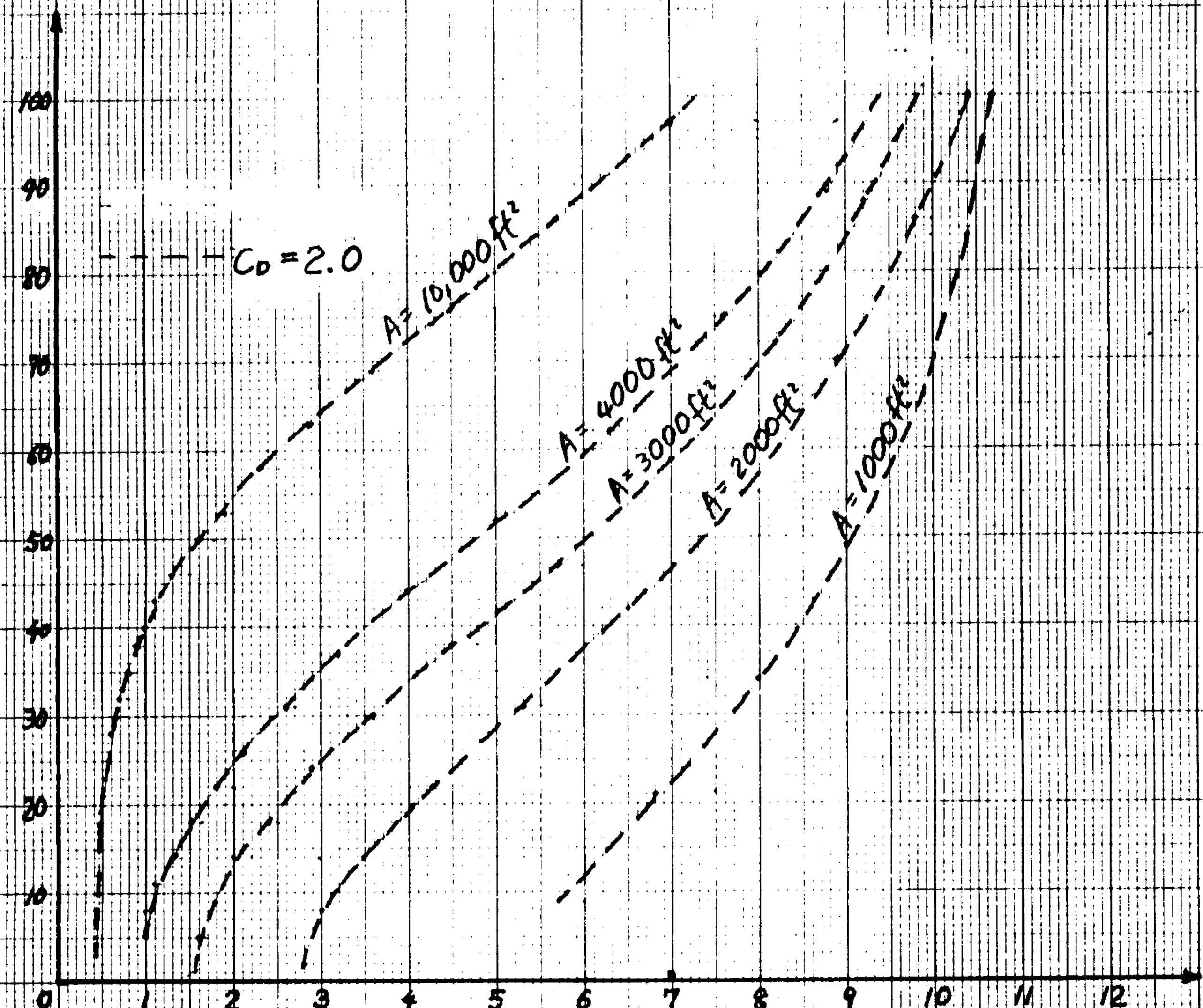


VELOCITY VS. ALTITUDE, $C_D = 1.0$



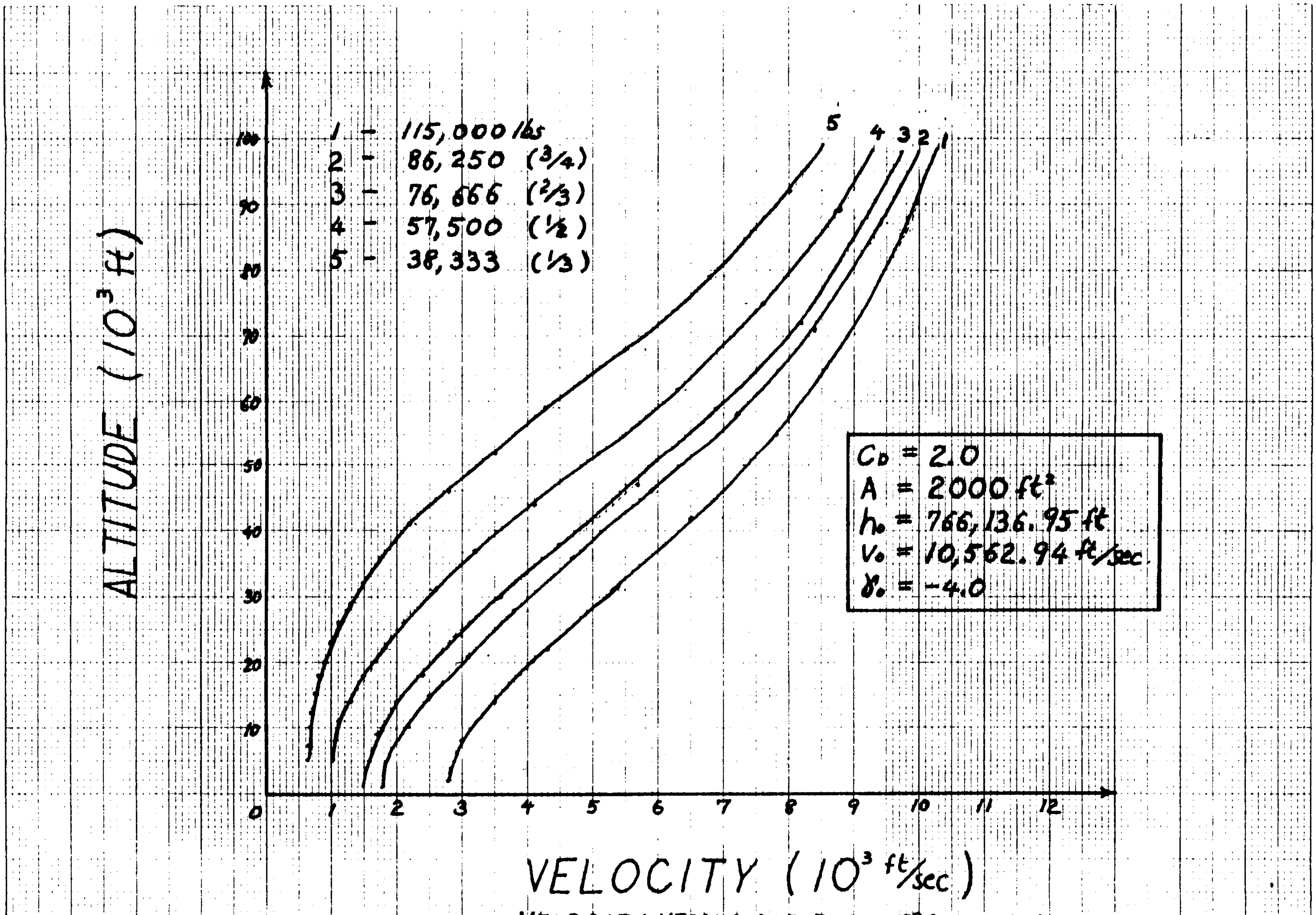
VELOCITY VS. ALTITUDE, $C_D = 1.4$

FIGURE 4.3.3.1.3



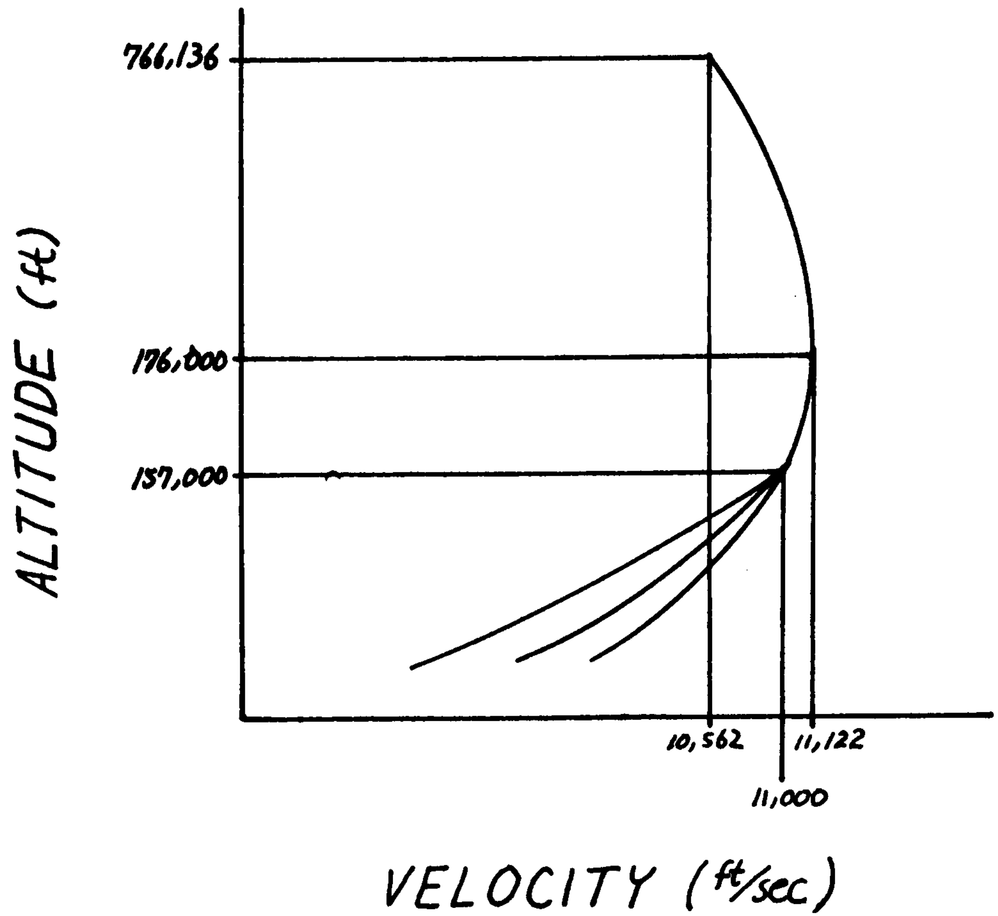
VELOCITY VS. ALTITUDE, $C_D = 2.0$

FIGURE 4.3.3.1.4

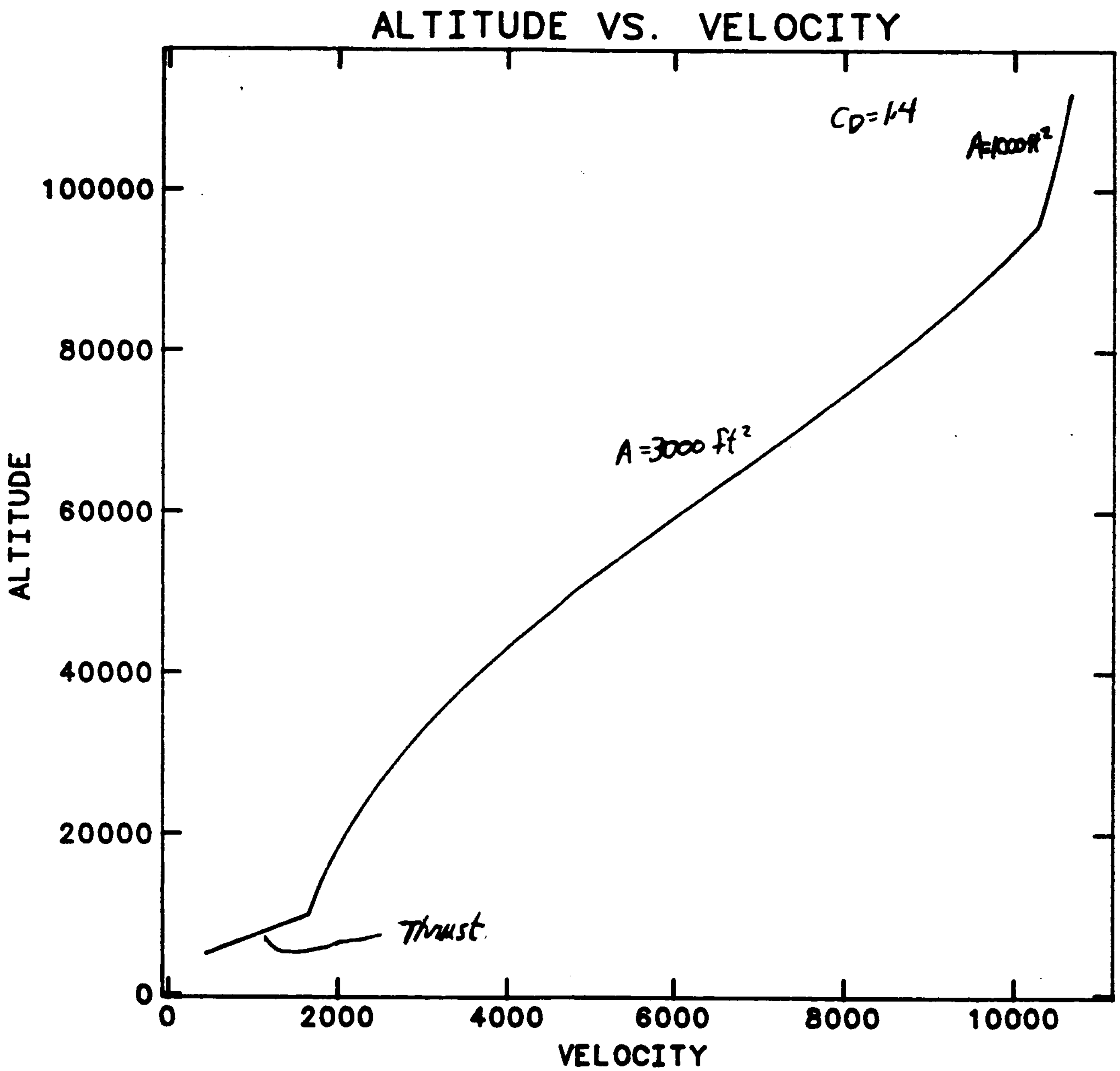


VELOCITY VERSUS ALTITUDE VERSUS MASS

FIGURE 4.3.3.1.5



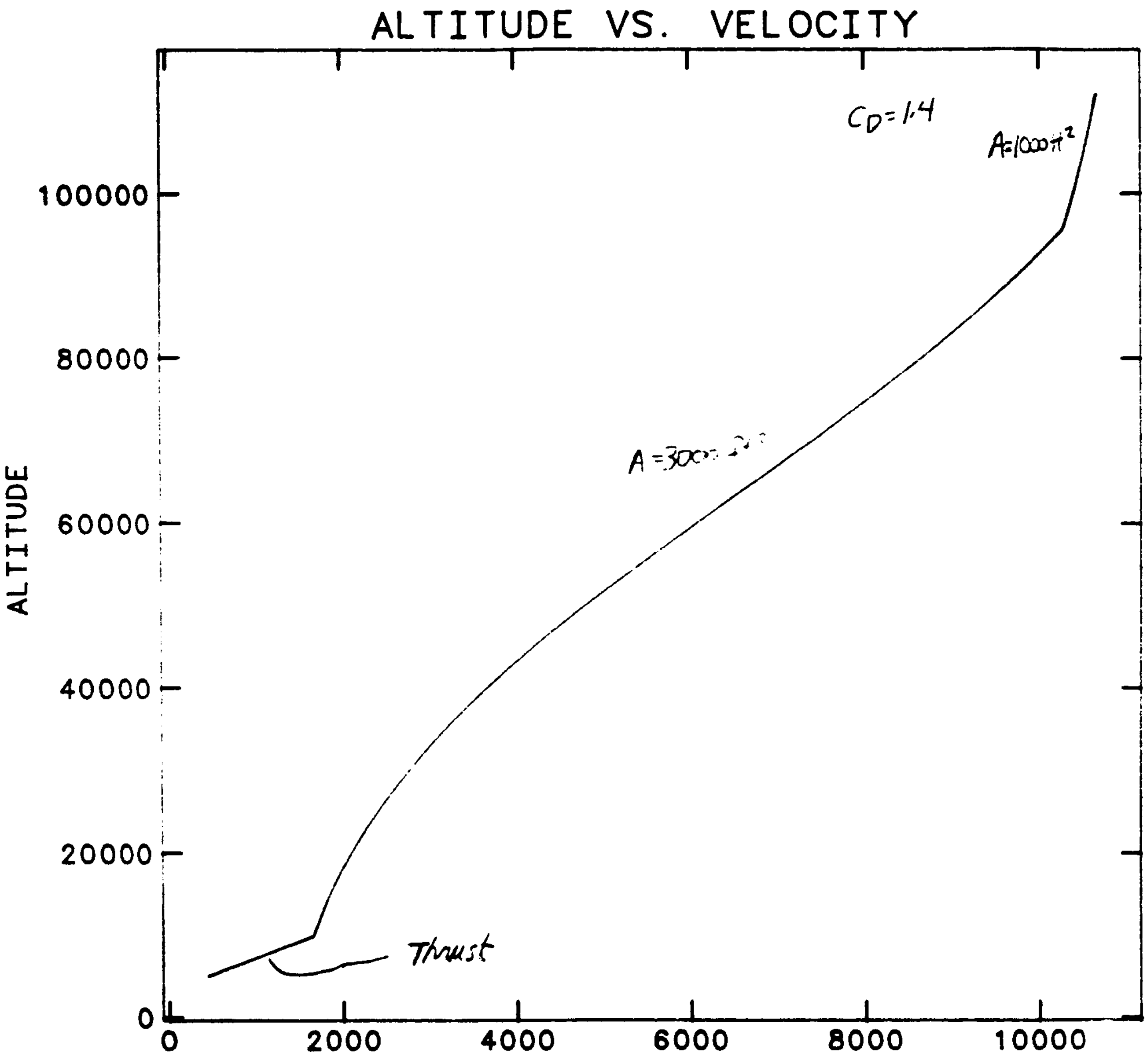
EFFECT OF ATMOSPHERE ON DESCENT PROFILE



DESCENT VELOCITY PROFILE INCLUDING THRUST

FIGURE 4.3.3.1.7

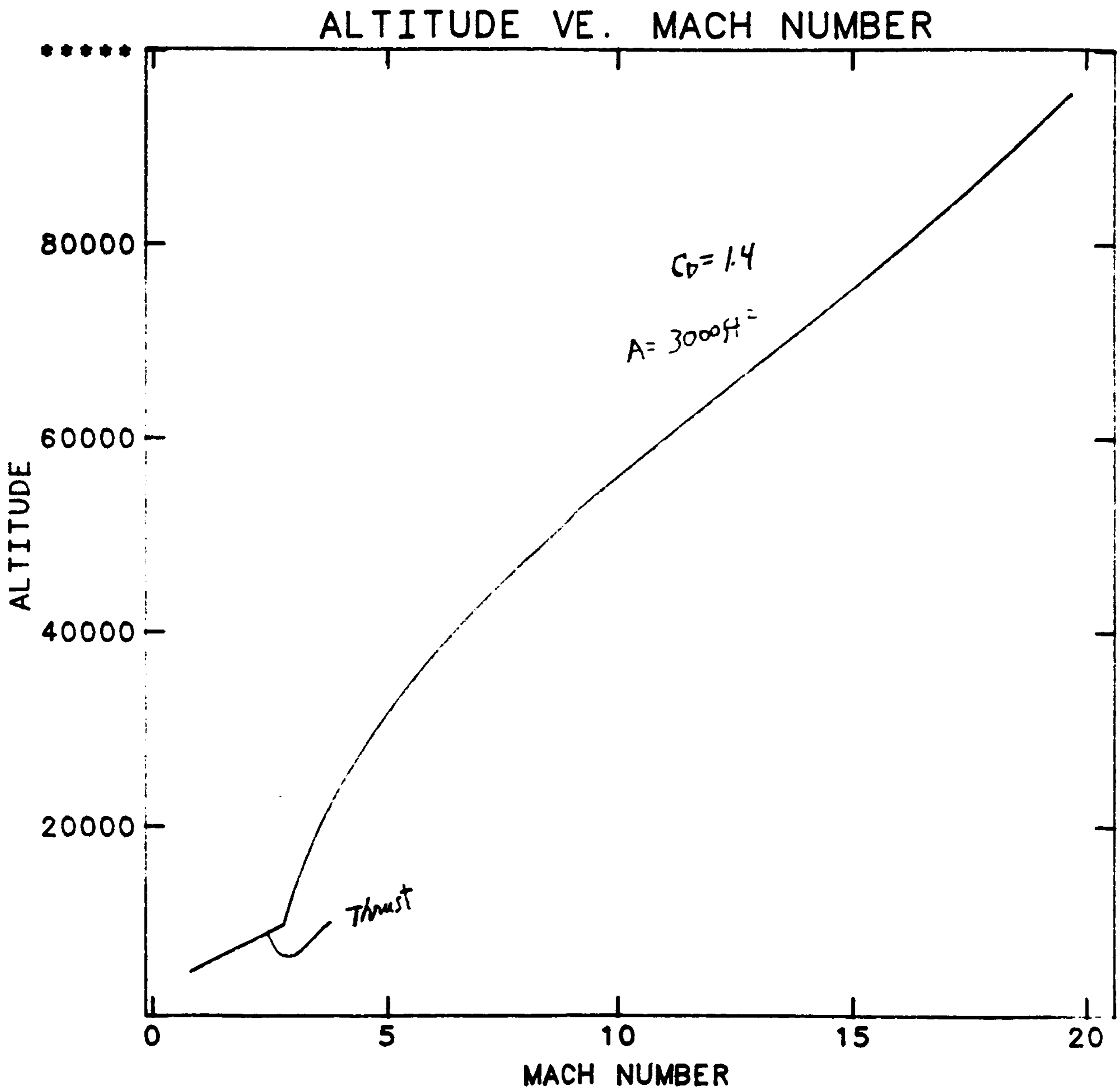
00.57.26.
27 APR 85



DESCENT VELOCITY PROFILE INCLUDING THRUST

FIGURE 4.3.3.1.7

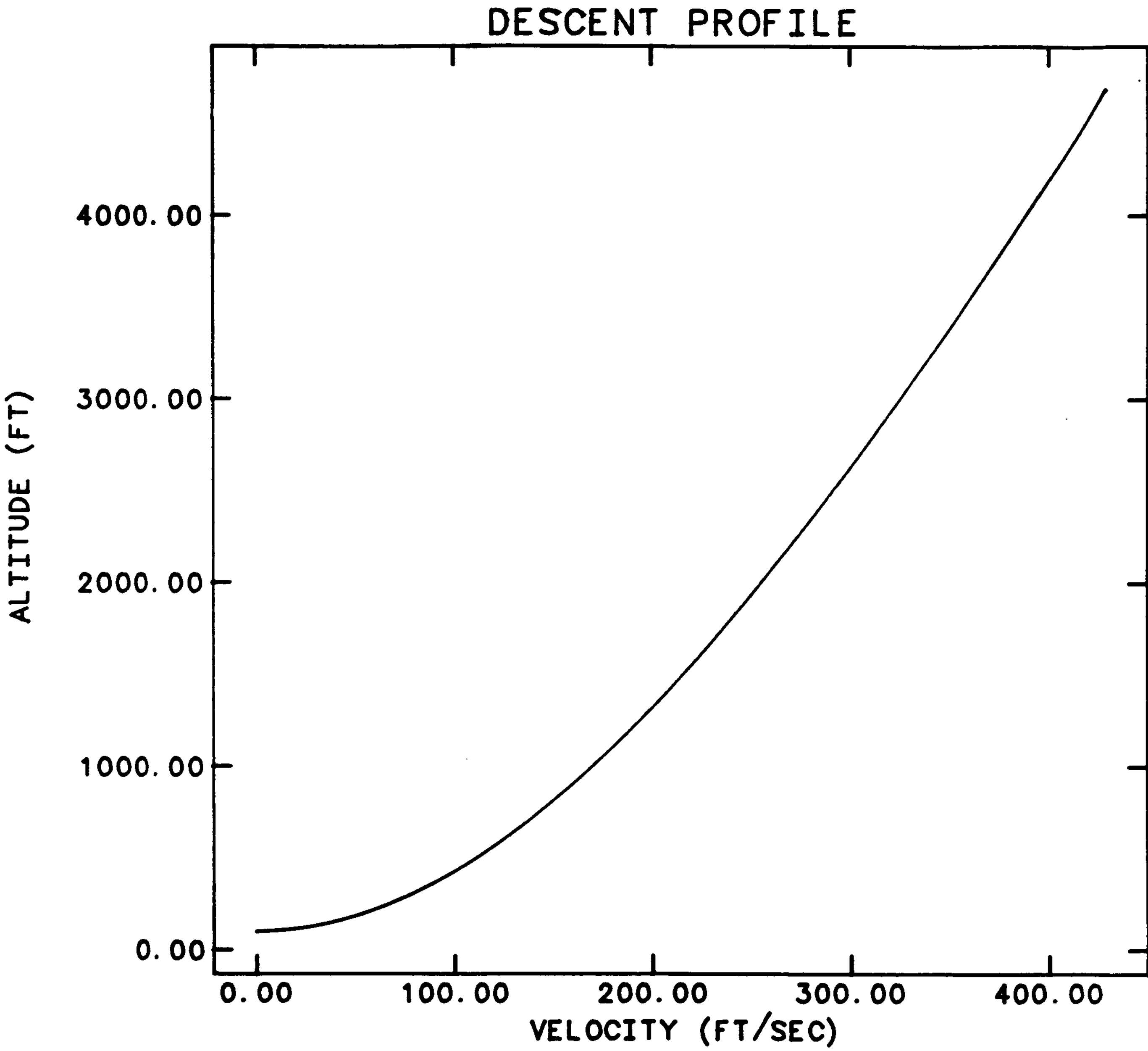
00.57.26.
27 APR 85



DESCENT MACH PROFILE INCLUDING THRUST

FIGURE 4.3.3.1.8

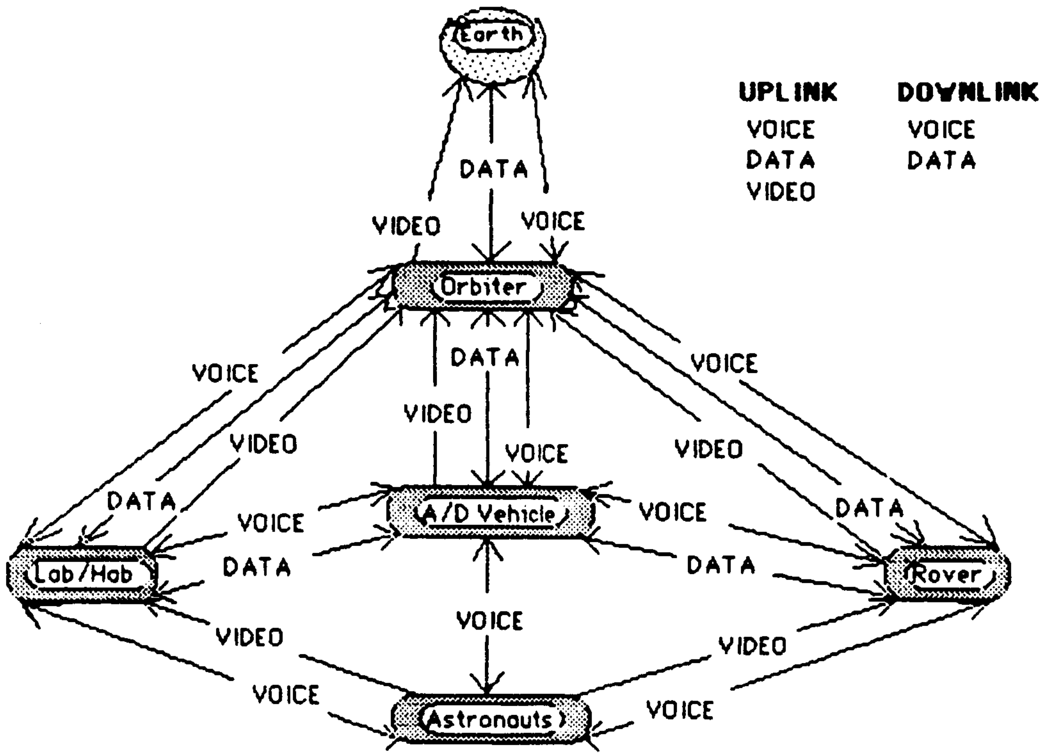
01.05.57.
27 APR 85



A/D VEHICLE DESCENT PROFILE

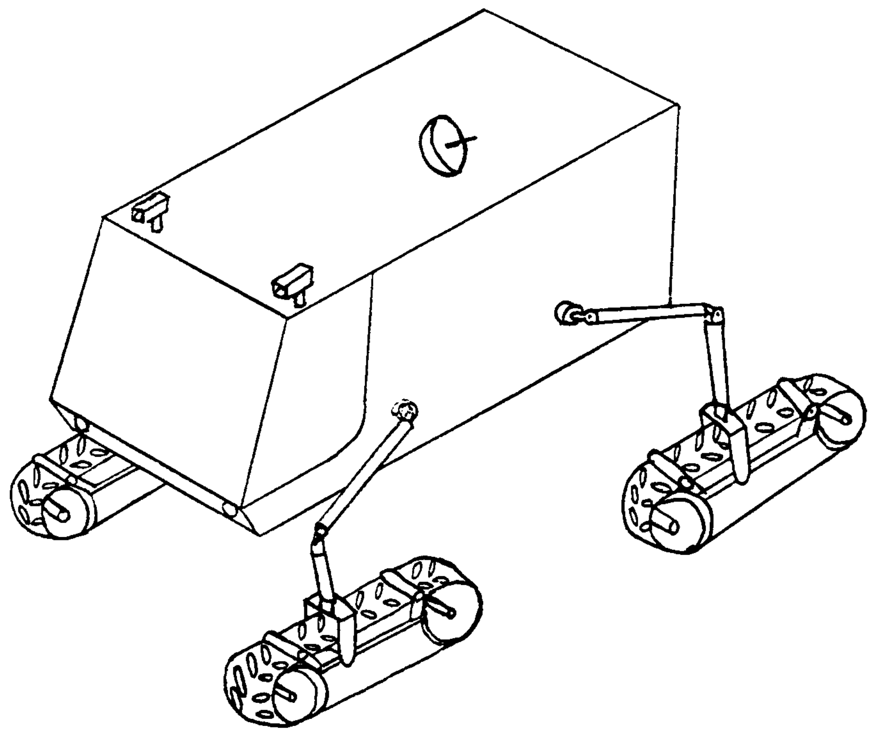
FIGURE 4.3.3.2.1

21.05.42.
16 APR 85



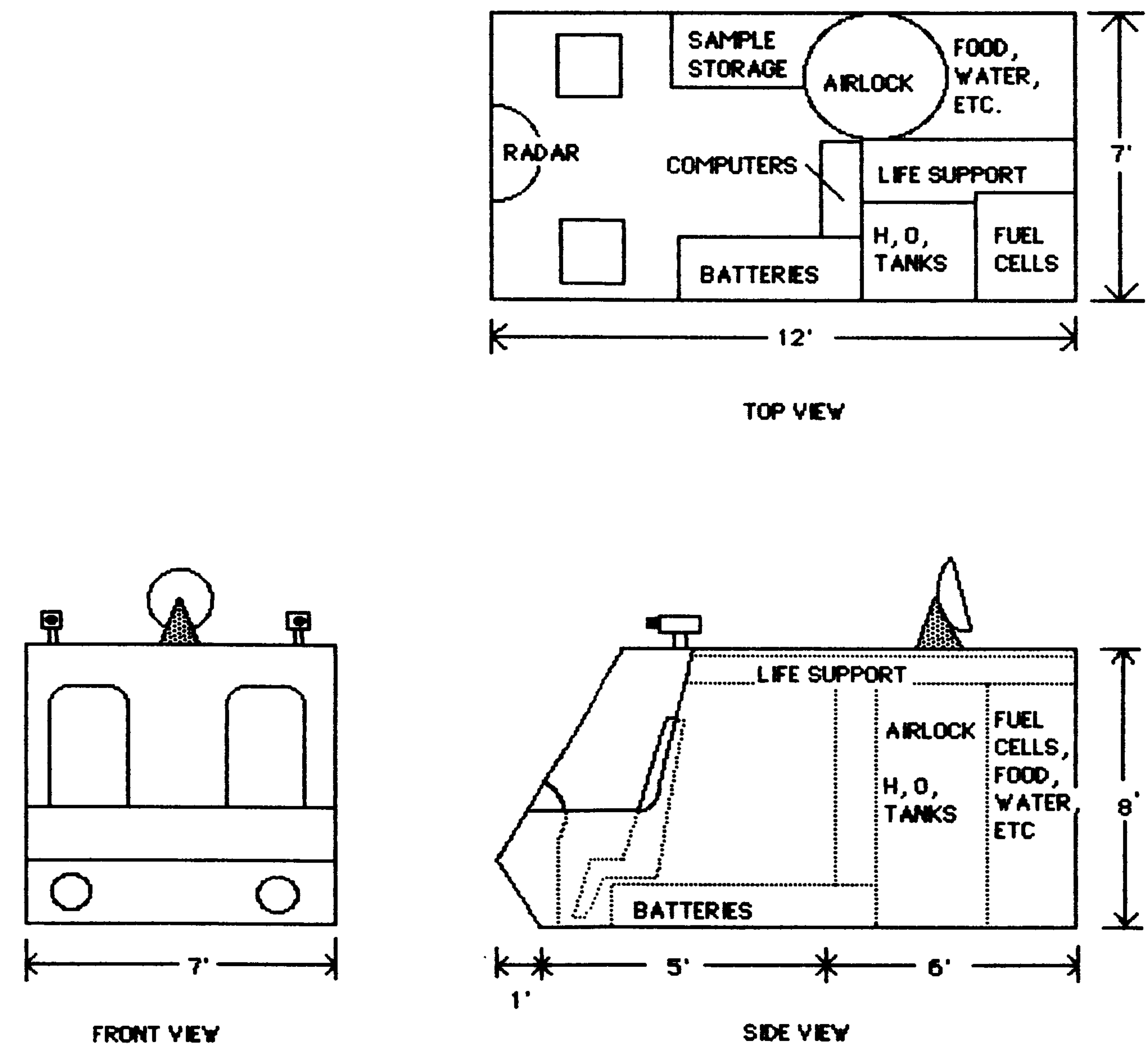
COMMUNICATION AND DATA LINKS

FIGURE 4.4.1



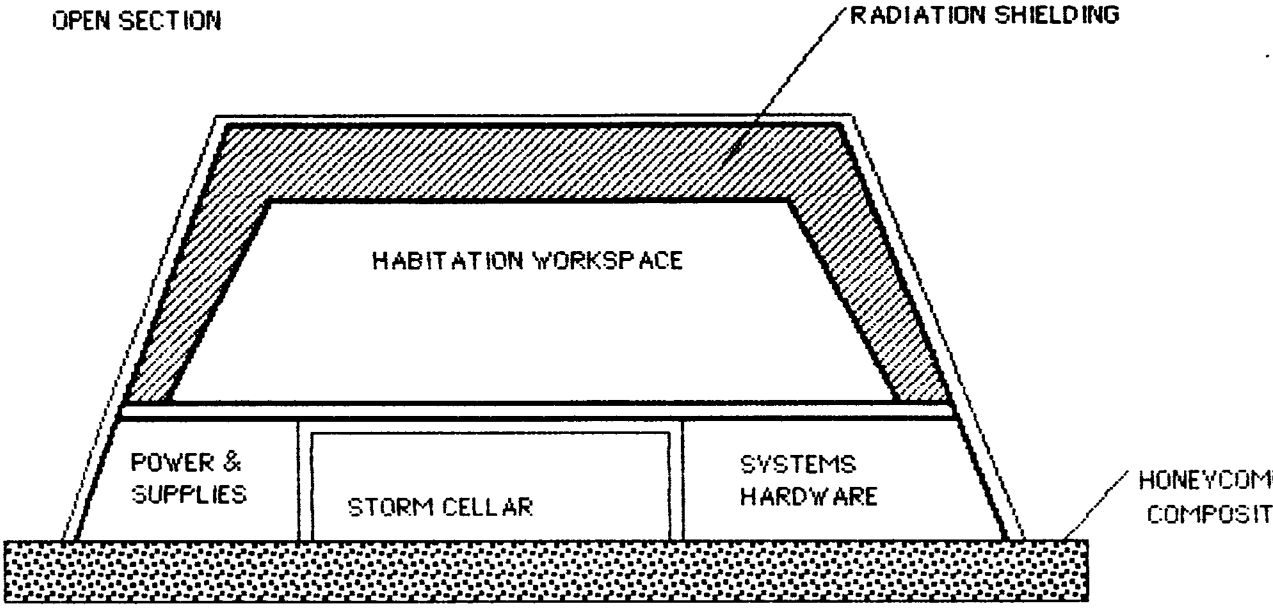
SHIRT-SLEEVE ROVER

FIGURE 4.4.2



VIEWS OF ROVER SUBSYSTEMS

FIGURE 4.4.3



HABITATION WORKSPACE

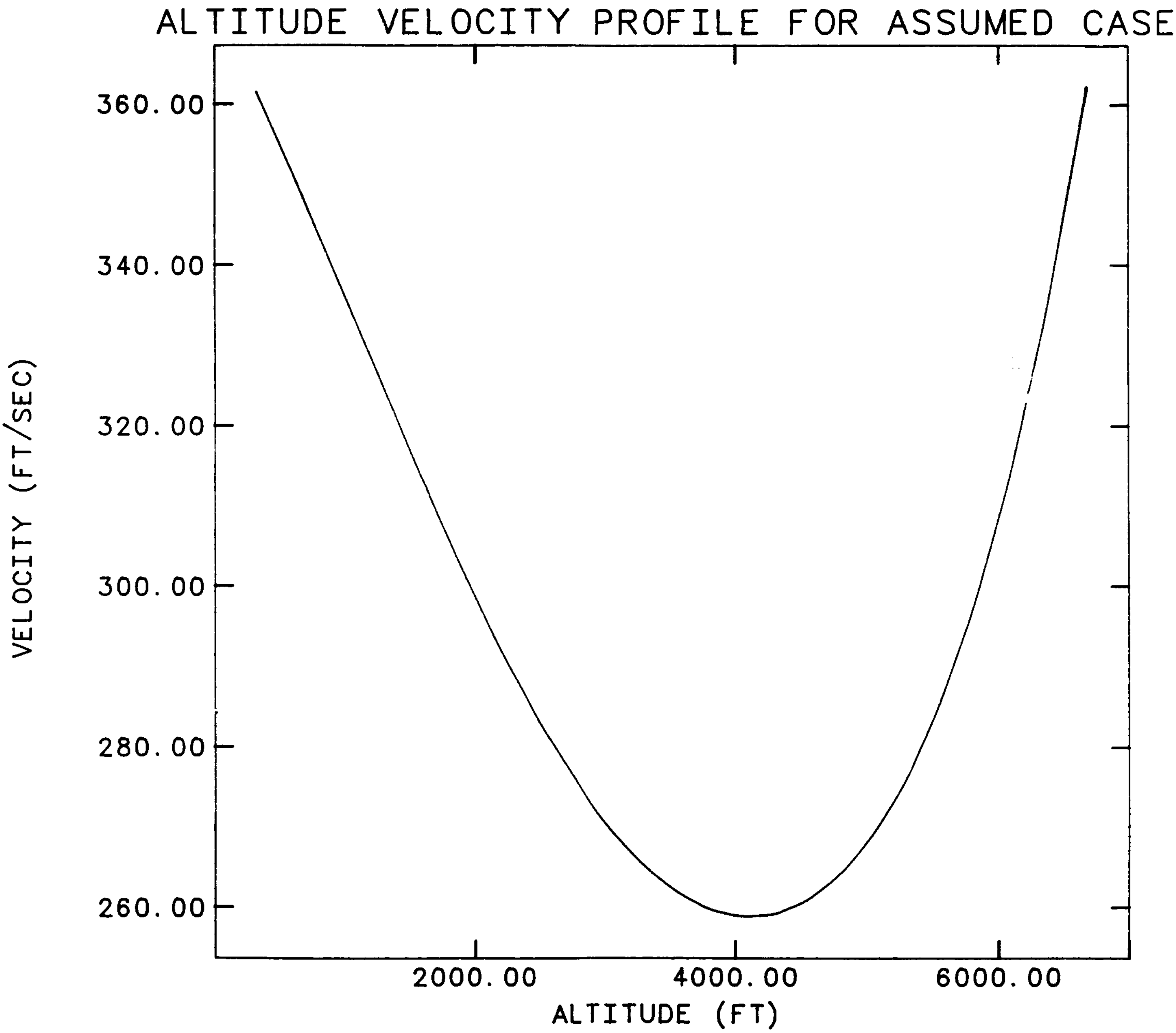
- 2 SLEEP AREAS
- CENTRAL SPACE
- DATA REDUCTION/STORAGE
- TELEMETRY/COMMUNICATIONS
- SCIENCE EQUIPMENT
- TOOLS
- FOOD STORAGE/PREPARATION
- HAND WASH/URINAL/CLOTHES WASH
- TOILET
- LIFE SUPPORT CONTROLS
- POTABLE WATER
- MAIN AIRLOCK
- SERVICE AIRLOCK

SUB WORKSPACE AREA

- LIFE SUPPORT SYSTEMS HARDWARE
- POWER GENERATION/STORAGE
- WASTE STORAGE
- ACTIVE THERMAL CONTROL HARDWARE
- STORM CELLAR
- SCIENCE EQUIPMENT STORAGE
- MODULAR SYSTEM REPLACEMENTS
- RADIATORS

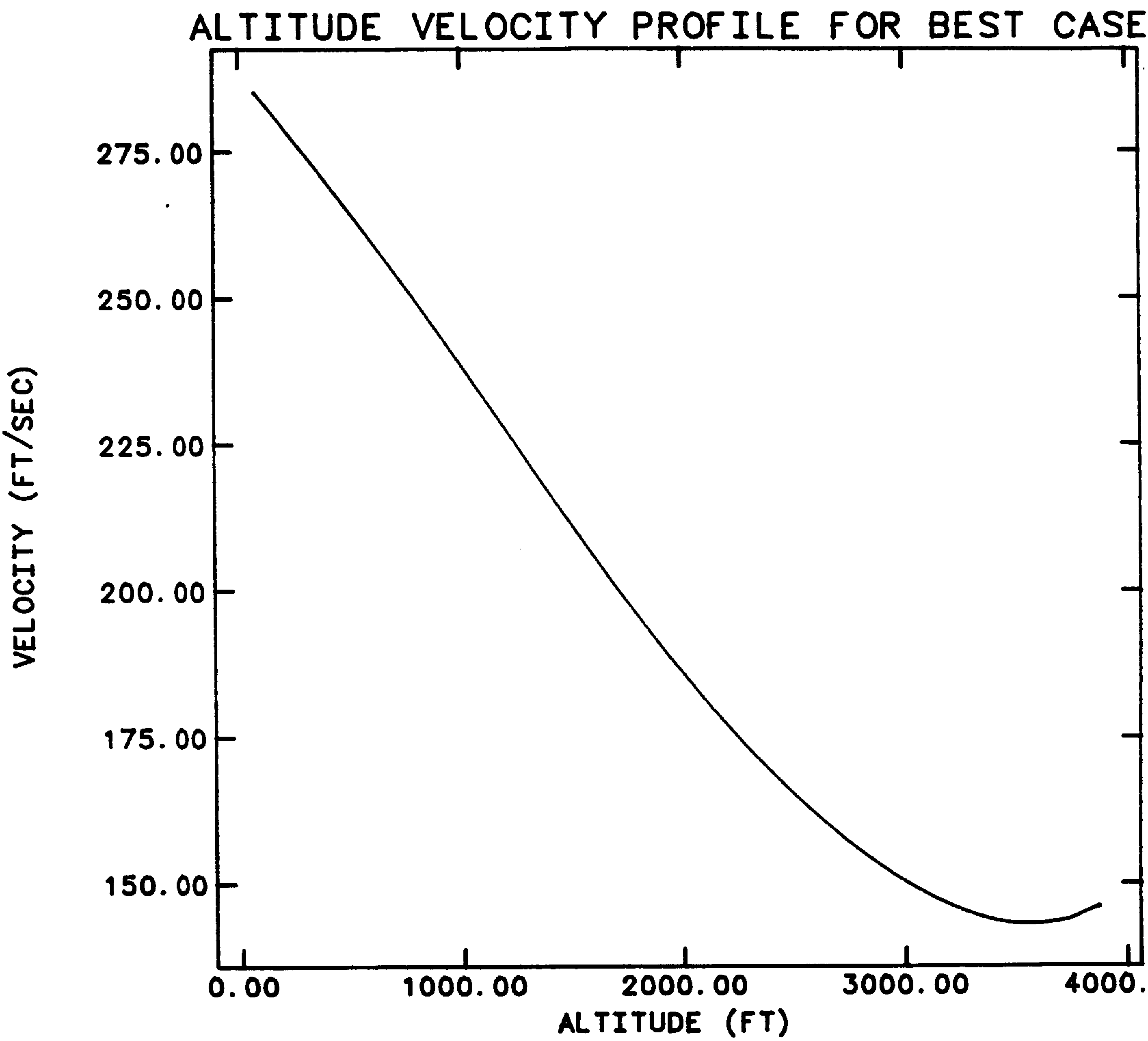
IMMOBILE LABORATORY / HABITAT

FIGURE 4.5.2.1



LAB/HAB VELOCITY VS. ALTITUDE (ASSUMED CASE)
FIGURE 4.5.4.1

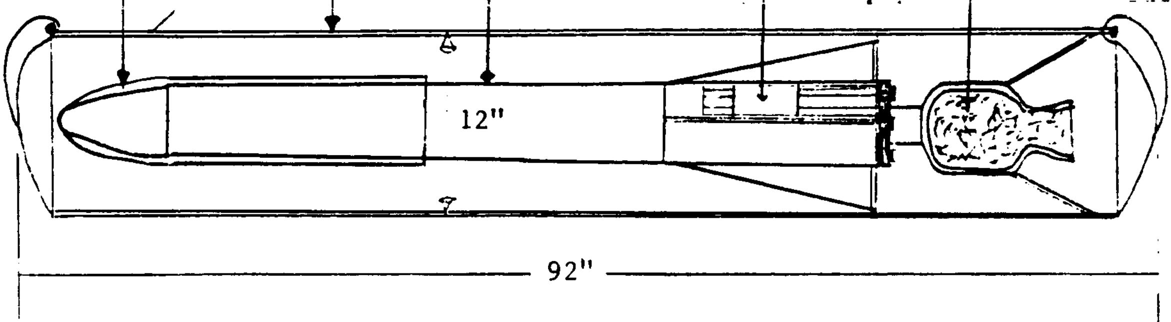
15.04.33.
29 APR 85



LAB/HAB VELOCITY VS. ALTITUDE (BEST CASE)

FIGURE 4.5.4.2

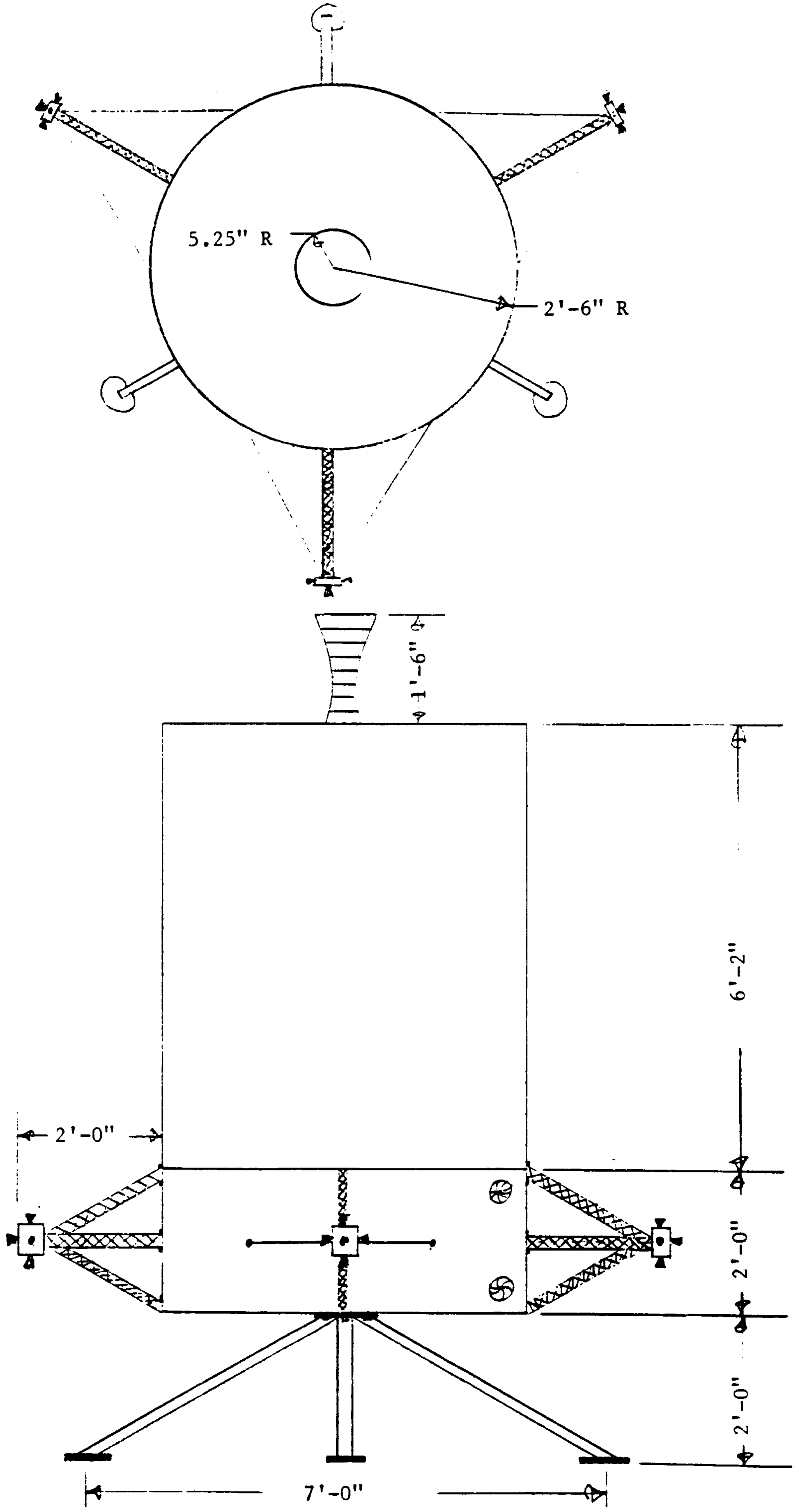
16.09.58.
01 MAY 85



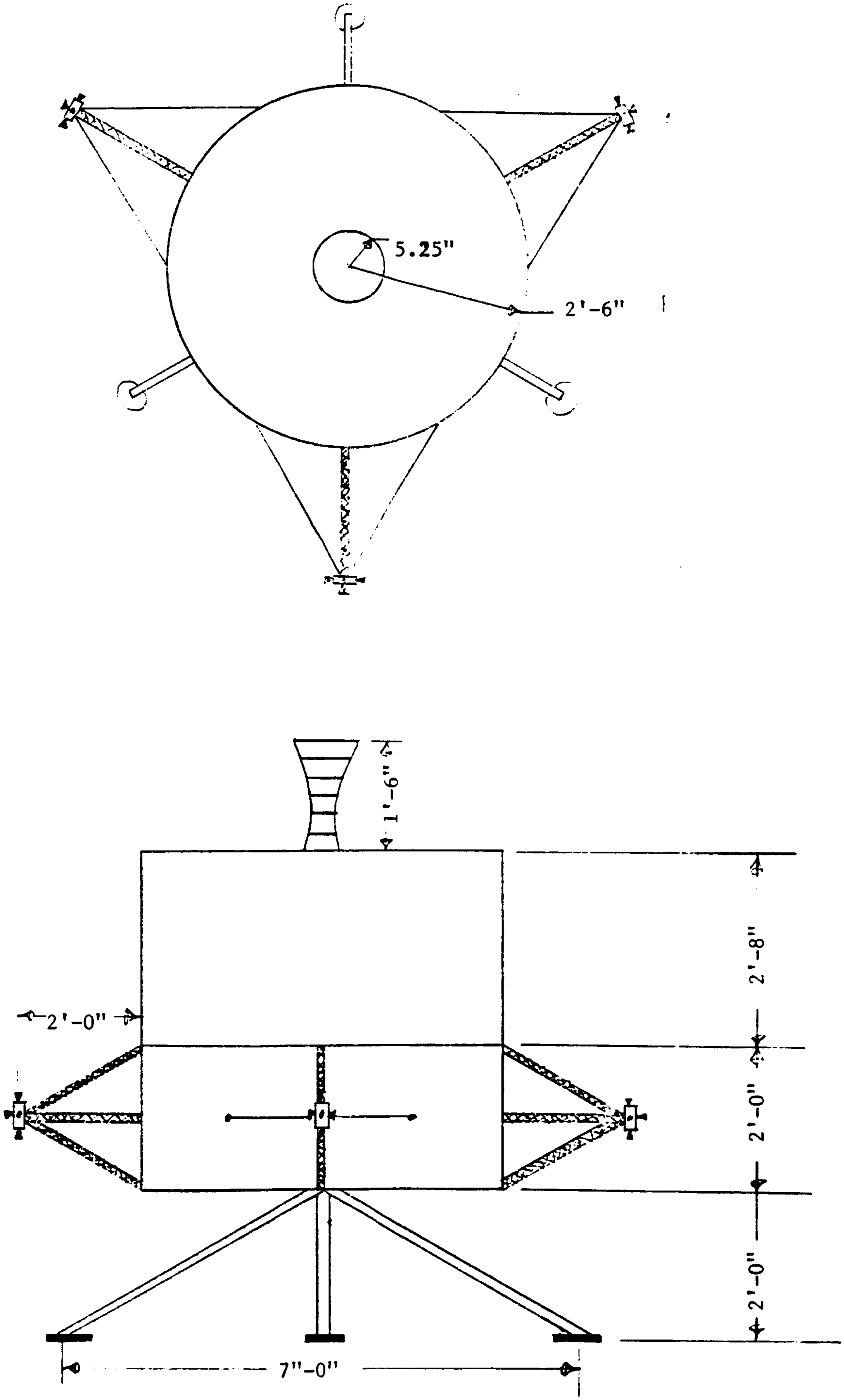
Maximum System Weight is 166 lbs.

Penetrator System and Launch Tube Assemblies

FIGURE 4.6.1.1



Soft Lander for Deimos
FIGURE 4.6.2 1



Soft Lander for Phobos

<u>FIGURE 4.6.6.[illegible]</u>

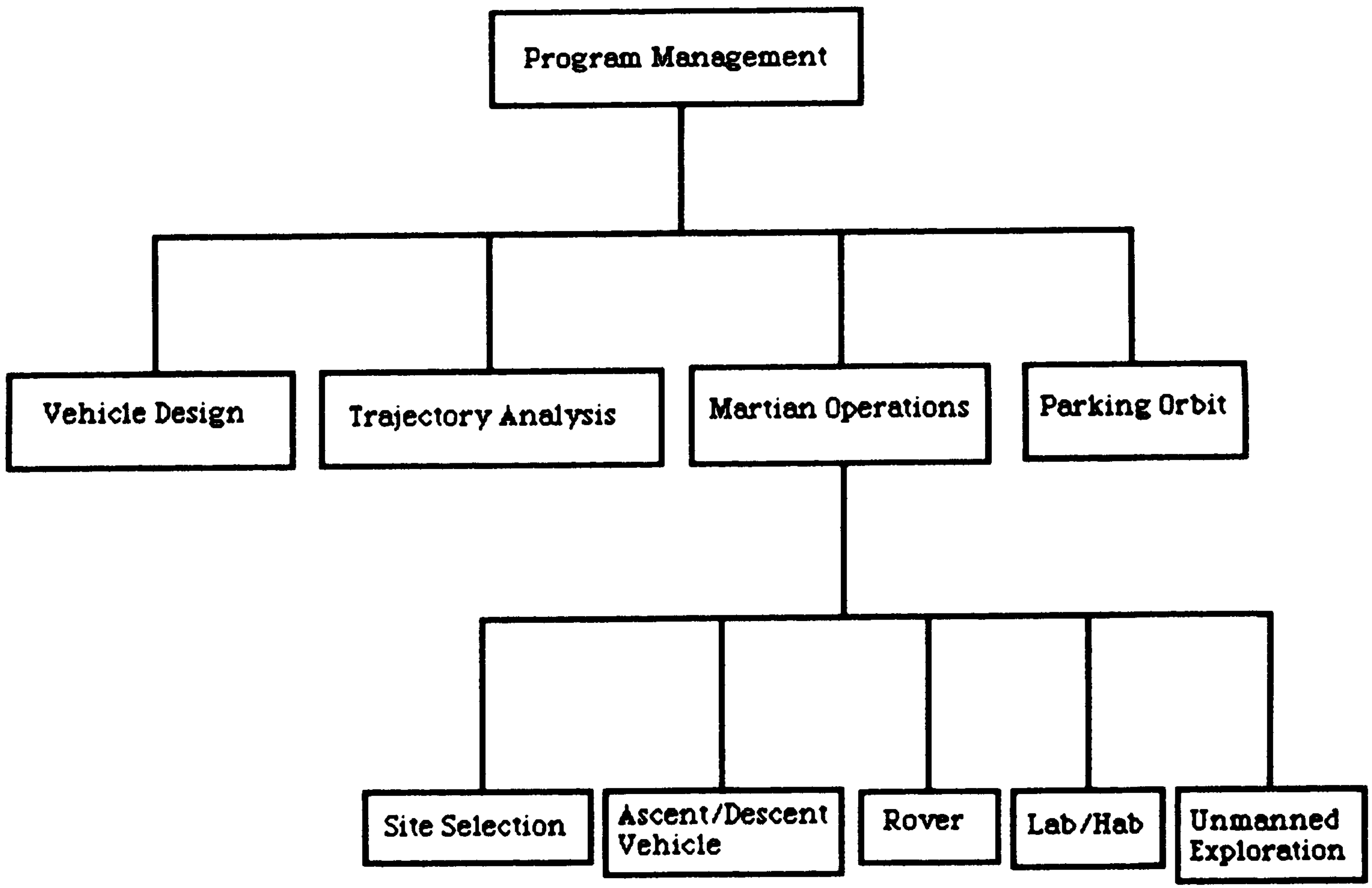


FIG. 5.1.1 ORGANIZATION STRUCTURE

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A. Esbach, Ovid W. and Mott Sunders, Handbook of Engineering Fundamentals, 3rd ed., New York; John Wiley and Sons, 1975, p482

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C. Bekker, M.G., Introduction to Terrian-Vehicle Systems, Ann Arbor: The University of Michigan Press, 1969, pp. 650-653

D. Space Shuttle Systems Summary, Space Systems Group, Rockwell International, SSV80-1, May 1980, pp. 172-173

APPENDIX 1: RADIATION SHIELDING

Developing radiation shielding to protect the crew from radiative energy has in the past, not been fully developed for long-duration space flights. Protection for short-duration flight came from arrangement of on-board equipment and from the required wall structure that was needed to protect from micrometeoroids. This gave sufficient radiation protection to meet mission dose constraints.

This method of protection is not sufficient for missions of extended periods in space. Material composition, proton cutoff energy, and density must be looked into to develop shield thicknesses and weights so that they may be optimized to conserve on overall weight and cost.

Using weight optimization equations from "Weight Optimization Methods in Space Radiation Shield Design," Journal of Spacecraft and Rockets, thicknesses and weights for the Lab/Hab and Rover were developed. The equation for the thickness is:

$$t = [F(\text{density}) * R(E)]/\text{density}$$

where

$$F(\text{density}) = \text{equivalence function}$$
$$= 1 - 5/3 * e^{(-1.386(\text{sqrt}(\text{density})))}$$

$$R(E) = \text{universal range function } (g/cm^2)$$
$$= 556 * \ln(1 + 5.48 * 10^{-6} (E * 1.8)) \quad (E \text{ is in MeV})$$

By using these equations, the required thicknesses for polyethylene, water and aluminum to provide adequate protection levels for a 100 MeV proton energy level were calculated. The maximum energy level for normal radiation activity (solar and cosmic rays) is approximately 100 MeV. The thicknesses are as follows:

polyethylene: t=0.24 ft (2.88 in)
water: t=0.24 ft (2.76 in)
aluminum: t=0.12 ft (1.45 in)

Any one of these materials is considered adequate to provide the necessary radiation protection.

The following method was used to calculate the thickness and shield weight for the Lab/Hab and Rover.

1) A combination of polyethylene and aluminum, or water and aluminum was used to conserve weight and to provide a structural shell.

2) The area of the shielded region was determined.

3) The volume to the shielding was determined.

4) The weight of the shield was calculated for each section.

5) Total shield weight was determined.

- Lab/Hab

1) Assume that the Lab/Hab has dimensions of 25 ft diameter and 8 ft high. The bottom 5 ft of the Lab/Hab is equipment, therefore it does not require shielding because it will be provided by the equipment that is stored under the floor level. Also assume the sides of Lab/Hab do not require as much shielding as the top.

Total surface area of top= 491 ft^2 (πR^2)

Total surface area of sides= 628 ft^2 ($2\pi Rh$)

2) Trade off some polyethylene for aluminum to provide a structural shell.

top: polyethylene: t=0.25 ft (3")
aluminum : t=0.042 ft (0.5")

sides: polyethylene: t=0.17 ft (2")
aluminum : t=0.042 ft (0.5")

3) Volume of shielding (t * surface area):

top: polyethylene: V=122.75 ft^3
aluminum : V=20.6 ft^3

sides: polyethylene: V=107 ft^3

aluminum : $V=27\ ft^3$

4) Calculated weight (V * density):

top: polyethylene: W=7050 lb
aluminum : W=3472 lb

sides: polyethylene: W=6145 lb
aluminum : W=1551 lb

Total Shield Weight For The Lab/Hab = 18,218 lbs

• <u>Rover</u>

1) Assume 12'x7'x8' Rover, total surface area to be shielded except the bottom, also assume sides do not require as much shielding.

Total surface area of top=84 ft^2 (LxW)

Total surface area of sides=304 ft^2 (2(LxH)+2(WxH))

2) Trade off some water for aluminum to provide a structural shell.

top: water: t=0.25 ft
aluminum: t=0.042 ft

sides: aluminum: t=0.125 ft

3) Volume of shielding (t * surface area):

top: water: $V=21\ ft^3$
aluminum: $V=3.53\ ft^3$

sides: aluminum: $V=36.5\ ft^3$

4) Calculated weight (V * density):

top: water: W=1311 lb
aluminum: W=595 lb

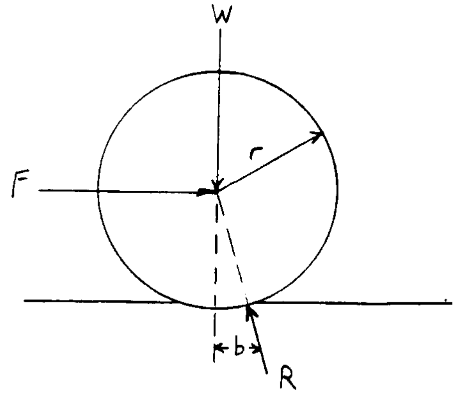
sides: aluminum: W=2279 lb

Total Shield Weight For The Rover = 4185 lbs

(This weight is a rough estimate. The front of the Rover will have to have some other type of shielding over the front window.)

Appendix 2

Rolling Friction



$Wb = Fr$ where $W = 3666.67\ lb_f$

$b = 0.1\ in.$

$r = 12\ in.$

$$F = \frac{Wb}{r}$$

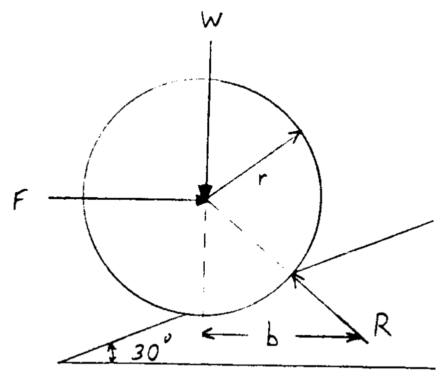
$$F = \frac{(3666.67\ lb)(0.1\ in)}{12\ in}$$

$$F = 30.55\ lb_f$$

b = Coefficient of Rolling Friction

R = Reaction Force

r = radius



$F = m g_m + \frac{Wb}{r}$ where g_m = Martian gravity

$F = \frac{W}{g_m} g_m + \frac{Wb}{r}$

$b = 0.12\ in$

$F = W\left(1 + \frac{b}{r}\right)$

$F = 3666.67\ lb_f\left(1 + \frac{0.12\ in}{12\ in}\right)$

$F = 3703.3\ lb_f$

Appendix 3

Power Requirements

$$\text{Power} = \text{Force} \cdot \text{velocity}$$

Transversing level ground at a maximum velocity of 10 mph,

$$P = F \cdot V$$

$$P = (30.55\ lb_f)(10\ mi/hr)\left(\frac{5280\ ft}{mi}\right)\left(\frac{hr}{3600\ s}\right)\left(1.356\ \frac{W}{\frac{lb_f \cdot ft}{s}}\right)$$

$$P = 607.6\ W$$

$$P = 0.6076\ kW$$

Transversing a 30° incline at selected maximum velocities,

$V = 2$ mph : $P = (3703.3\ lb)(2\ mi/hr)(5280\ ft/mi)\left(\frac{1}{3600}\ hr/s\right)\left(1.356\ \frac{W}{\frac{lb \cdot ft}{s}}\right)$

$P = 14.73\ kW$

$V = 3$ mph : $P = 22.09\ kW$

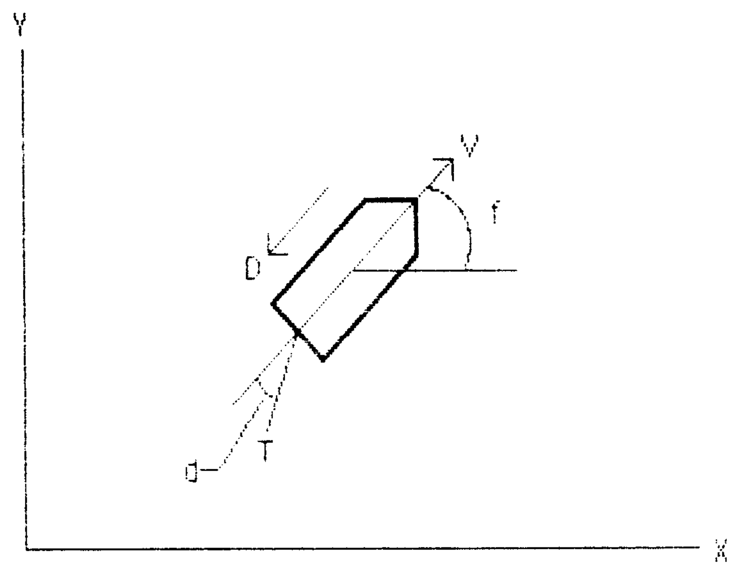
$V = 4$ mph : $P = 29.46\ kW$

$V = 5$ mph : $P = 36.825\ kW$

APENDIX 4: ASCENT PERFORMANCE USING OPTIMIZATION

An analysis of the ascent performance of the A/D Vehicle was attempted using an optimization subroutine. The objective was to minimize time of flight, using this as the optimization variable. Through the use of this subroutine, the performance required for the A/D Vehicle could be determined. This information would then be used in deciding the fuel to be used for the ascent vehicle.

The ascent was modelled using the coordinate system as shown below.



Five equations of motion were chosen:

State vector	Derivative vector
x	$x' = v \cos f$
y	$y' = v \sin f$
v	$v' = T/m \cos d - D/m - g \sin f$
f	$f' = v/y \cos f - g \cos f/v + T \sin d/mv + L/mv$
m	$m' = -T/I_{sp}g_c$

where x = x-position
y = y-position
v = velocity
f = flight path angle
m = mass of vehicle
T = thrust
d = gimbal angle of engine

I_{sp} = specific impulse
g_c = 32.2 lb_m ft/ lb_f sec^2

Given the initial conditions, the program integrates the trajectory of the A/D Vehicle and then performs the optimization. For the initial analysis, terms with drag or lift would be omitted from the equations.

In addition to these conditions, several constraints were put on the problem. The problem was set up to stop when the velocity and the y-distance equalled that of the parking orbit and when the flight path angle f = 0. Another factor in the equations was the gimbal angle of the engines. In the first set of runs, the gimbal angle was to be unconstrained during the flight. In the second set of runs, the gimbal angle would be held equal to or less than 20 degrees.

In the first set of runs with unconstrained gimbal angles, the constraints did not converge to their proper values. This same problem occurred in the second set of runs as well. A simpler set of equations was used with the same constraints. This set of runs showed all of the gimbal angles to be 20 degrees. This indicated that adjustments needed to be made to the constraints. There was not enough time, however, to complete the analysis. We recommend that more analysis be done using a simpler model to try to get some satisfactory results. After that point is reached, the complexity of the model can be increased to achieve more realistic results.

1/3

DATA SHEET

ENGINEERING CAREER ASSISTANCE CENTER
The University of Texas at Austin
Ernest Cockrell, Jr., Hall 2.4
Austin, Texas 78712

PERSONAL DATA

Name: Greeson, David S.
(Last, First, Middle Initial)

College Address: 7117 Woodhollow #613 Austin, TX 78731
(Street, Zip Code)

College Phone: (512) 343-0088

Permanent Address: 232 Star Hereford, TX 79045
(Street, City, State, Zip Code)

Permanent Phone: ( ) 806 364-1096

U.S. Citizen ☒ Yes ☐ No | Permanent Visa ☐ No. | Social Security No. [redacted] | Date Available June 10, 1985

GOALS

Brief Statement of Career Interests and/or Description of Work Desired
(Graduate Students List Thesis/Dissertation Title) An engineering job which is directly related to the civilian space program in any of the following areas: mission planning, guidance, navigation, and control, orbital mechanics, systems design, and computer applications. I will consider any aerospace engineering job offer, but my real interest is in the Space Station and Space Shuttle Programs.

Location Preference
Open

COLLEGE INFORMATION

NAME AND LOCATION OF COLLEGES ATTENDED (In Reverse order)	DATES From	DATES To	DEGREE EXPECTED BS,MS,PhD	GRADUATION DATE	FIELD OF STUDY (1) Major (2) Option	G.P.A. (1) Overall (2) Major	GRADE BASIS
The University of Texas Austin, Texas 78712	09/80	5/85	B.S.	5/18/85	1 Aerospace Eng. 2 Orbital Mech.	1 2.97 2 2.85	A = 4.0
					1 2	2	A =
					1 2	1 2	A =

College Honors, Professional Societies, Fraternities, and Activities
Sigma Gamma Tau, American Inst. Aeronautics Astronautics

G.P.A. by Year: ____ 1st ____ 2nd ____ 3rd ____ 4th ____ 5th

% College Expenses Earned: 65%
How Earned: Summer, Part-time, Co-op jobs

EMPLOYMENT INFORMATION

WORK EXPERIENCE (In Reverse Order) (Names and Addresses of Employers)	JOB TITLE	HOURS/ WEEK	DATES EMPLOYED From	DATES EMPLOYED To
International Business Machines 1322 Space Park Drive, Houston,TX	Engineering Co-op *(digital autopilot	40	6/84	9/84
same as above	same as above (digital auto.)	40	9/83	1/84
same as above	same as above (Shuttle flight software)	40	1/83	5/83
United States Air Force, Kelly AFB San Antonio, TX	same as above (structures)	40	6/82	8/82
Geosource Incorporated, Exploration Midland, TX	field crew worker, cable repair	60	6/81	8/81

GENERAL INFORMATION

References (Name, Title and Business Address—Preferably Business Reference, Past Supervisors, or Faculty)
Charles Hooks, Staff Programmer, IBM 1322 Space Park Drive, Houston, TX 77058

James Cannon, Quality Manager, Kelly AFB, San Antonio, TX 78241

Other Information (Community Activities, Hobbies, and Interests. Also, Organizations Outside of College)
University Baptist Church, Karate, Judo, Skiing, Racqetball, four wheel drive vehi

I authorize the Engineering Placement Office to release this information and/or a copy of my transcript to prospective employers and also I affirm that the above is true and correct.

Signature: David Greeson   Date: 15 Jan 85

*See reverse and attached resume for more information on prior work experience.

DAVID S. GREESON

UNIVERSITY ADDRESS (until June 1985)	PERMANENT ADDRESS
7117 Woodhollow #613	232 Star
Austin, TX 78731	Hereford, TX 79045
AC 512/343-0088	AC 806/364-1096

CAREER OBJECTIVES

Seeking a position as an engineer in the aerospace industry with involvement in the U.S. space program. Would prefer a company with opportunities to earn increased technical and managerial responsibilities.

EDUCATION

Bachelor of Science in Aerospace Engineering, May 1985; The University of Texas at Austin
Current Overall GPA: 2.97
Current Major GPA: 2.85
Earned 65% of college expenses

WORK EXPERIENCE

Co-op Engineer. International Business Machines, Federal Systems Division. Houston, Texas. Summer 1984, Fall 1983, Spring 1983. Produced computer software in connection with the Space Shuttle and Space Station programs applied to the areas of onboard guidance, navigation and control, local area network modelling, database management, and flight simulation. Analyzed data to determine conformance to NASA specifications and satisfaction of engineering requirements. Documented software for end users and for proposals to be made to NASA.

Co-op Engineer. United States Air Force, Kelly Air Force Base. San Antonio, Texas. Summer 1982. Designed aircraft structural repairs and parts-specific machine tool fixtures. Performed stress calculations, materials substitutions, aircraft part inspection and measurement, and documented repairs made to aircraft. Also investigated industry composite material manufacturing methods and wrote a report detailing what new equipment would have to be purchased and what new training would be required to begin manufacturing composite material spare parts at Kelly Air Force Base.

Seismic Field Crew. Geosource Incorporated, Petty-Ray Geophysical Division. Midland, Texas. Summer 1981, Summer 1980. Worked as a crew clerk in charge of recording expenses and transmitting payroll and sesmic data to main office. Perfomed repair to and built seismic cables including shop testing. Drove a cable truck to pick-up and deliver cables, drove a vibrator vehicle to provide seismic energy source, and served as the vibrator mechanic's assistant in the field.

HONORS AND ACTIVITIES

Sigma Gamma Tau Aerospace Engineering Honor Society, Dean's Honor Roll, National Dean's List, American Institute of Aeronautics and Astronautics. Cockrell Foundation scolarship, Air Force Reserve Officer Training Corps four year scholarship (since dropped).

INTERESTS

Personal computers (programming, bullentin board services, and graphics), future space exploration, science fiction and science fact, and scale modeling. Athletic interests include snow and water skiing, racquetball, Karate, Judo, and running. Off-road four wheel drive vehicles and touring in the Rocky Mountains.

REFERENCES

Available upon request.

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Phone: (512) 345-9277

EDUCATION

University of Texas at Austin Austin, Texas Major: Aerospace Engineering GPA in Major: 3.96/4.0	Fall 1980 BS expected Spring 1985

Relevant Courses:
- Structural Dynamics
- Aerospace Structural Analysis
- Aerospace Materials Laboratory
- Design and Testing of Aerospace Structures
- Orbital Mechanics

University of Houston Houston, Texas	Summer 1984 Summer 1983
San Jacinto College Houston, Texas	Spring 1982 Fall 1982
Round Rock High School Round Rock, Texas	September 1977-May 1980

WORK EXPERIENCE

University of Texas at Austin Dr. R. R. Craig Laboratory Assistant	Spring 1985

Worked in Structural Dynamics Laboratory. Assisted Dr. Craig in analysis of large space structures.

Intel Corporation Austin, Texas Contract Employee (Part-time) Documentation Consultant	September 1984-December 1984

McDonnell Douglas Technical Services
Houston, Texas
Co-op Engineer

Attitude and Pointing Analysis Task	Summer 1984

Communication assessments and attitude timeline analysis for several spacelab missions.

Propulsive Consumables Analysis Task — Summer 1983, Fall 1982

Developed propulsive consumables requirements and maintained orbiter mass properties data for several different missions.

Attitude and Pointing Analysis Task — Spring 1982

Navigation software verification and orbiter attitude analysis for solar impingement studies.

University of Texas at Austin — Spring 1984
Dr. R. R. Craig
Laboratory Assistant/Grader (Part-time)

Worked in Structural Dynamics Laboratory. Assisted Dr. Craig in analysis of large space structures.

Gibsons Products Incorporated — October 1978-December 1981
Austin, Texas
Department Manager (30 hours/week)

HONORS AND ACTIVITIES

Tau Beta Pi - National Engineering Honor Society
Sigma Gamma Tau - National Aerospace Engineering Honor Society
Engineering Honor Roll
American Institute of Aeronautics and Astronautics (AIAA) - Treasurer

PERSONAL DATA

Birthdate: June 10, 1962	Health: Excellent
Height: 5'10"	Marital Status: Married
Weight: 150 lbs.	Hobbies: Camping, hiking and softball

REFERENCES AVAILABLE UPON REQUEST

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DATA SHEET

ENGINEERING CAREER ASSISTANCE CENTER
The University of Texas at Austin
Ernest Cockrell, Jr., Hall 2.4
Austin, Texas 78712

PERSONAL DATA

Name: Long, Walter R., III
(Last, First, Middle Initial)

College Address: 2529 S. Lamar, #59, 78704
(Street, Zip Code)

Permanent Address: 807 S. 7th St., Kenedy, Tx. 78119
(Street, City, State, Zip Code)

College Phone: (512) 444-9176

Permanent Phone: 512 583-3342

U.S. Citizen ☒ Yes ☐ No

Permanent Visa ☐ No.

Social Security No. [redacted]

Date Available: June 1,1985

GOALS

Brief Statement of Career Interests and/or Description of Work Desired
(Graduate Students List Thesis/Dissertation Title)

1st Choice: Flight Testing--Would eventually like to test the airworthiness of newly designed aircraft

2nd Choice: Orbital Mechanics

Location Preference: No Preferences

COLLEGE INFORMATION

NAME AND LOCATION OF COLLEGES ATTENDED (In Reverse order)	DATES From	DATES To	DEGREE EXPECTED BS,MS,PhD	GRADUATION DATE	FIELD OF STUDY (1) Major (2) Option	G.P.A. (1) Overall (2) Major	GRADE BASIS
The University of Texas Austin, Texas 78712	9-80	5-85	BS	5-85	1 Aero. Engineer. 2 N/A	1 3.25 2	A = 4.0
University of Houston at Clear Lake, Houston, Tx.	1-84	5-84	N/A	N/A	1 N/A 2 N/A	3.0 2 N/A	A = 4.0
San Jacinto College Pasadena, Tx. 77505	1-82	8-83	N/A	N/A	1 N/A 2 N/A	1 4.0 2 N/A	A = 4.0

College Honors, Professional Societies, Fraternities, and Activities:
Tau Beta Pi, Sigma Gamma Tau, Phi Eta Sigma, Co-op Program

G.P.A. by Year: 3.696 1st 3.428 2nd 3.333 3rd 3.295 4th 3.252 5th

% College Expenses Earned: App. 50%

How Earned: Co-op Program

EMPLOYMENT INFORMATION

WORK EXPERIENCE (In Reverse Order) (Names and Addresses of Employers)	JOB TITLE	HOURS/ WEEK	DATES EMPLOYED From	DATES EMPLOYED To
NASA--Johnson Space Center Houston, Tx. 77058	Co-op Student	40	1-82	8-84
Ragsdale Aviation, Inc. Austin, Tx.	Line Boy	16	6-82	8-82

GENERAL INFORMATION

References (Name, Title and Business Address—Preferably Business Reference, Past Supervisors, or Faculty)

Dick Jacobs, Co-op Coordinator, University of Texas, Austin, Tx. 78712 (ECJ, Rm. 2.4

Bill Morris, ECLSS Manager, NASA--JSC, Houston, Tx. 77058 (Bldg. 7A, Mail Code: EC2)

Other Information (Community Activities, Hobbies, and Interests. Also, Organizations Outside of College)

Flying, Jogging, Martial Arts, Sailing, SCUBA Diving

I authorize the Engineering Placement Office to release this information and/or a copy of my transcript to prospective employers and also I affirm that the above is true and correct.

Signature: W. Russ Long

Date: 1-18-85

BRYAN E. MARTIN
Bachelor of Science in Aerospace Engineering
May 1985

PRESENT ADDRESS
2311 Lafayette, Suite B
Austin, Texas 78722
(512) 476-9394

PERMANENT ADDRESS
661 Red River
Lewisville, Texas 75067
(214) 221-2376

OBJECTIVE: To find a challenging opportunity to utilize my engineering and computer background in an Aerospace company that offers stability and growth potential.

EDUCATION: 1980-1985 The University of Texas at Austin
Major: Aerospace Engineering
Concentration: Computer Applications
* Passed EIT Exam - Spring 1984

EXPERIENCE:

FLIGHT CONTROL - Texas Instruments, Dallas, TX
Title: Software Technician (Summer 1984)
Duties: Using top-down design, wrote Jovial code to couple the vertical and horizontal algorithms in a terrain following program.

COMPUTER GRAPHICS - UT Aerospace Department, Austin, TX
Title: Computer Programmer (1983-1984)
Duties: Generated plotting routines for wind tunnel test data.

WIND TUNNEL TESTS - Balcones Research Center, Austin, TX
Title: Undergraduate Research Assistant (Summer 1983)
Duties: Assisted in aerodynamic testing; built structure to stabilize wind tunnel exhaust.

MANAGEMENT POSITION - Sunbelt Energy Systems, Dallas, TX
Title: Crew Supervisor (December 1983)
Duties: Supervised commercial installment of insulation. Developed new installation method and trained crew to implement that method, resulting in substantial savings for the company.

RADAR AND WEAPON SYSTEMS - UT Naval ROTC, Austin, TX
Title: Officer Candidate (1980-1982)
Duties: Learned basics of Naval weaponry and detection systems; recorded/analyzed missile-target flight path data.

* Earned 100% of college expenses

HONORS AND ACTIVITIES:

American Institute of Aeronautics and Astronautics
- Committee Chairman (1983-1984)
- President (1984-1985)

TSPE and SEDS: member
Church Leadership: Home Group Leader (1982-1984)
Scholarships: NROTC (2 years), UPC (5 years)

DATA SHEET

ENGINEERING CAREER ASSISTANCE CENTER
The University of Texas at Austin
Ernest Cockrell, Jr. Hall 2.4
Austin, Texas 78712

PERSONAL DATA

NAME Last	First	Middle	Social Security Number
Metteauer	Michael	Scott	[redacted]

Present Address (Street, City, State, Zip) 904 W. 21st #114, Austin, Texas 78705	Present Phone 512-472-7410
Permanent Address 1102 Augusta, Lufkin, Texas 75901	Permanent Phone 409-634-4883

GOALS

Brief Statement of Career Interests and/or Description of Work Desired I am seeking both summer and permanent employment in aerospace engineering.	Check One: U.S. Citizen [X] Permanent Visa [ ] Student Visa [ ]
Location Preference Open	Date Available 5/85 and 1/86

COLLEGE INFORMATION

NAME AND LOCATION OF COLLEGES ATTENDED	DATES From	DATES To	DEGREE EXPECTED BS,MS,Ph.D	GRADUATION DATE	FIELD OF STUDY (1) Major (2) Option	GPA (1) Overall (2) Major	GRADE BASIS
THE UNIVERSITY OF TEXAS Austin, Texas 78712	9/81	5/84	B.S.	Dec 85	1. Aerospace 2.	1. 3.561 2. 3.979	A = 4.0
					1. 2.	1. 2.	A =
					1. 2.	1. 2.	A =

Thesis and/or Dissertation Title

G.P.A. by Year: ____ 1st ____ 2nd ____ 3rd ____ 4th ____ 5th

College Honors, Professional Societies, Fraternities, and Activities
Student Engineering Council, Sigma Gamma Tau, Eng. Honor Roll, AIAA TBΠ

% College Expenses Earned	How Earned
40	National Merit, Cockrell, and Texas Exes Scholarships College Scholar

EMPLOYMENT INFORMATION

WORK EXPERIENCE (Names and Addresses of Employers)	DESCRIPTIVE JOB TITLE	HOURS/ WEEK	DATES EMPLOYED From	DATES EMPLOYED To

GENERAL INFORMATION

References (Name, Title, and Business Address - Preferably business reference, past supervisors, or faculty)
Dr. Anthony M. Bedford, Professor, U.T. Dept. of Aerospace Engineering
Dr. Victor Szebehely, Professor, U.T. Dept. of Aerospace Engineering

Other Information (Community Activities, Hobbies, and Interests. Also, organizations outside of college)
Methodist Youth Fellowship, piano, racquetball, jazz music

I authorize the Engineering Placement Office to release this information and/or a copy of my transcript to prospective employers and also I affirm that the above is true and correct.

Signature Michael Scott Metteauer Date 9/14/84

Ernesto Moralez, III

University Address
(until May 1985)
1910 Ridgemont
Austin, Texas 78723
512/928-2365

Permanent Address
918 South 17th
Kingsville, Texas 78363
512/592-4835

CAREER OBJECTIVE - Applied research and development in areas of stability and control, flight testing, and flight simulation as applied to aerospace vehicles.

EDUCATION - The University of Texas at Austin
B.S. Aerospace Engineering, May 1985

Technical Area of Interest: Flight Mechanics

Overall GPA: 3.8/4.0
Major GPA (ASE/EM): 3.8/4.0

EXPERIENCE - Summer 1984
Undergraduate research assistant, Department of Aerospace Engineering and Engineering Mechanics, The University of Texas, Austin. Assembled and brought up to operational status a T-38 jet aircraft simulator donated by the U.S. Air Force. Helped maintain, repair, and operate the T-38 and the Link GAT-1 flight simulators in the Flight Simulation Laboratory.

Spring 1984
Engineering Co-op, Flight Dynamics and Controls Branch, National Aeronautics and Space Administration (NASA) Ames Research Center, Moffett Field, California. Developed computer code to generate time histories of the linearized math model of the YAV-8B Harrier prototype along with computer code to graphically compare the linearized math model response and the nonlinear math model response. Developed the code on a Digital VAX 11/780 and gained proficiency with the VAX/VMS operating system.

Summer 1983
Engineering Co-op, Thermochemical Test Branch, NASA Johnson Space Center, Houston, Texas. Involved in the testing of power and pressure systems. Test manager for two tests which entailed the following responsibilities: preparing contractor job orders, initiating procurements, writing test plans, monitoring test stand build-up and test article preparation, preparing test procedures, conducting the test, analyzing data, and writing test reports.

Fall 1982
Engineering Co-op, Aircraft Operations Division, NASA Johnson Space Center, Houston, Texas. Responsible for the data reduction of flight data tapes for the Shuttle Training Aircraft. Prepared a users guide for the computer graphics system which generated time histories of the reduced flight data. The graphics system consisted of a data-linked ROLM 1666 minicomputer and Tektronix 4054 Graphics Computer.

Summer 1982
Undergraduate Research Assistant, The University of Texas Balcones Research Center, Austin, Texas. Maintenance and repair of the 5'x7' subsonic wind tunnel. Assisted in testing of drag cone configurations for Tracor, a local aerospace company. Became familiar with the data acquisition system based on a Hewlett-Packard 9825 computer.

1981- present
Free lance photographer. Aerial, wedding, musical groups. Documentary photos for a Houston-based oil drilling company, cover photo work for local publications company, and full page pictorial photograph for the 1982 University of Texas yearbook, the Cactus.

AWARDS/SCOLARSHIPS - HONORS

Tau Beta Pi, Sigma Gamma Tau, Joe C. Walter, Jr. Endowed Presidential Scholarship (1982-83), T.U. Taylor Foundation Endowed Presidential Scholarship (1984-85), Rockwell International Company Scholarship (1983-84), Honor Roll of the College of Engineering (1982-1984), Engineering Scholar (1982-1984), College Scholar (1983), National Dean's List (1981-1984).

ACTIVITIES -

American Institute of Aeronautics and Astronautics (AIAA), photographer for AIAA, Tau Beta Pi, and the Vector, a publication of the Student Engineering Council. Member of the Cactus yearbook staff (1983), NASA Ames photography club.

REFERENCES -

Will be furnished upon request.

JONETTE STECKLEIN

BACHELOR OF SCIENCE IN AEROSPACE ENGINEERING
MAY 1985

PRESENT ADDRESS
2201 Willow Creek #179
Austin, TX 78741
(512) 440-7060

PERMANENT ADDRESS
9237 Midvale Drive
Shreveport, LA 711]
(318) 686-5162

OBJECTIVE

To obtain a challenging position involving high performance aircraft design or space exploration and support. My experience and interests lend themselves to work with a broad perspective, specifically, vehicle configuration, system organization, and mission requirements.

EDUCATION

BS ASE, May 1985 - The University of Texas at Austin
Overall GPA: 3.8/4.0
Earned 100% of college expenses

ENGINEERING EXPERIENCE

Lockheed Missiles & Space Company; Austin, TX

Systems Engineer — Summer 1984
Wrote a system test plan and a subcontract statement of work. Performed a requirements traceability study.

Software Engineer — Summer 1983
Assisted in writing a Program Design Specification and a Data Base Design Document. Used assembler source listings to write a functional description of the software.

Program Office Engineer — Fall 1982
Researched procedures to determine compliance with configuration management and quality assurance policies. Implemented a new correspondence tracking system.

Federal Emergency Management Agency; Thomasville, GA

Engineering Technician — Summer 1982/Summer 1981
Surveyed buildings and determined the structures' ability to withstand nuclear weapons effects. Performed ventilation analyses.

HONORS AND AWARDS

Southwest Region Outstanding Aerospace Engineering Student; Tau Beta Pi (National Engineering Honor Society); Mortar Board (Leadership Honorary Society); National Merit Scholarship; Engineering Honor Roll; Engineering Scholar

ACTIVITIES

President, Sigma Gamma Tau (Aerospace Honor Society)
American Institute of Aeronautics and Astronautics; Student Engineering Council; U. T. Competitive Dance Team

WOH PENG FUN
P.O. BOX 7118
AUSTIN, TEXAS 78713
(512) 471-4909

OBJECTIVE: Applied reseach in Aerodynamics with special interest in computational fluid mechanics.

EDUCATION: B.S.A.S.E. 1985. The University of Texas, Austin.

GPA in major 3.8 on a 4.0 scale.

Financed 75% of tuition and fees through work experience.

EXPERIENCE: Summer 1984 to Present -- Undergraduate Research Assistant in Aerodynamics.

January 1984 to May 1984 -- Grader for the Department of Aerospace Engineering and Engineering Mechanics.

September 1983 to December 1983 -- Grader for the Mathematics Department.

March 1983 to August 1983 -- Dormitory Cafeteria Worker.

HONORS: Engineering Scholar 1984 and 1985.
College Scholar 1984 and 1985.
National Dean's List 1983 and 1984.
Chancellor's Scholarship 1981.
Sigma Gamma Tau (National Honorary).

INTERESTS: Table Tennis, basketball, swimming, jogging, single engine airplane flying.

REFERENCES: Available upon request.